Chapter 7
Mission Design

Abstract During the entire phase of space transportation system mission, from lift-off till satellite injection, various constraints and requirements applicable not only to the mission but also to vehicle systems, ground systems, range safety and tracking systems are to be satisfied. Considering these constraints and requirements, an optimum feasible trajectory has to be designed to meet the mission requirements. The mission design process involves the utilization of the available energy for realizing the defined orbital mission by devising suitable strategies of directing the energy along the suitable path and sequencing the energy addition process. Optimum mission design strategies have to be arrived at to achieve the maximum performance, ensuring the defined mission under nominal and off-nominal flight environments and system parameter dispersions. The trajectory shaping satisfying vehicle loads and radio visibility for continuous tracking coverage during ascent phase are other essential parts of mission design. There are number of constraints like thermal loads on the spacecraft, the vehicle subsystems during ascent phase and jet plumes of reaction control system thrusters interacting with the spacecraft. The passivation requirements of the final stage after spacecraft separation are to be carefully worked out. Another important aspect of mission design is to finalize the flight events/sequences which generate various commands to separate the stages and to initiate the subsequent flight events. In such complex systems close interactions among various disciplines exist, and the mission design requires several iterations. In this chapter all these aspects of mission design are discussed in detail and various activities involved in mission design process explained. The mission design strategies and importance of the same for the vehicle design process are included. Mission requirements, constraints, design and analysis aspects and trajectory design constraints during various phases of trajectory are presented. Mission sequence design considerations and all other mission-related studies like satellite orientation requirements for multiple satellite launch and passivation requirements to ensure the safety of the spent stage are also highlighted.

Keywords Mission design • Mission specifications • Trajectory design • Lift-off studies • Load relief • Gravity turn trajectory • Wind biasing • Thermal design • Mission sequence • Stage passivation • Vehicle tracking and range safety

7.1 Introduction

Once a space transportation system and the associated vehicle subsystems are designed to provide the necessary energy to position the satellite into its specified orbit, it is essential to carry out a detailed integrated mission design to ensure that all the constraints and requirements of the mission, vehicle, range and tracking systems are satisfied. Mission design is the process of utilizing the available energy for realizing the defined orbital mission by devising suitable strategies of (1) directing the energy along the suitable path, (2) sequencing the energy addition process and (3) way of utilizing the energy by shaping the energy addition process. Optimum mission design strategies have to be arrived at to achieve the maximum performance, ensuring the defined mission under nominal and off-nominal flight environments and system parameter dispersions. While achieving the above performance requirement, the design has to satisfy the constraints and capabilities of the systems such as (1) vehicle systems (structural loads), (2) vehicle subsystems (control power plant, navigation, guidance and autopilot and thermal) and (3) mission constraints arising from lift-off, range safety and tracking.

The mission design comprises of studying the requirements and associated constraints in totality and carrying out detailed analysis and studies, to ensure that the mission objectives are completely fulfilled. The mission constraints vis-à-vis the objectives specified, launch vehicle capabilities and range safety–related aspects during the ascent phase are to be studied. The trajectory shaping/design satisfying vehicle loads, other constraints and radio visibility for continuous tracking coverage during ascent phase are other aspects which need consideration. There are also constraints like thermal loads on the spacecraft and the vehicle subsystems during ascent phase and jet plumes of reaction control system (RCS) thrusters of upper-stage control system interacting with the spacecraft. The passivation requirements of the final stage after spacecraft separation are to be carefully examined. Important element of mission design is the requirement for finalizing the flight events/sequences which generate various commands including the real-time decision for commands using on-board computer (OBC) to separate the stages and to initiate the subsequent flight events.

In such a complex system like STS close interactions among various disciplines exist, and the mission design requires several iterations. Therefore the design not only needs a systems approach for meeting all the defined objectives but also demands clear domain knowledge of all involved disciplines.

In this chapter all aspects of mission design are discussed in detail. Initially various activities involved in mission design process are explained. The mission design strategies and importance of the same for the vehicle design process are included. Mission requirements, constraints, design and analysis aspects and trajectory design constraints during various phases of trajectory are presented. Mission sequence design considerations and all other mission-related studies like satellite orientation requirements for multiple satellite launch and passivation requirements to ensure the safety of the spent stage are also highlighted.

7.2 Mission Design Activities

During different phases of STS mission, from lift-off till satellite injection and further passivation of the vehicle, various constraints and requirements applicable not only to the mission but also to vehicle systems, ground systems, range safety and tracking systems are to be satisfied. Considering these constraints and requirements, an optimum feasible trajectory has to be designed to meet the mission requirements. The total process comprising of the above activities is called mission design. The major tasks involved in the mission design process are given in Fig. 7.1.

The criticalities of each activity, the requirements and constraints imposed by these activities, which impact on the mission performance are summarized below and further explained in detail in the subsequent sections.

(i)   The lift-off sequence and vertical rise time have to be designed to ensure clear lift-off while maximizing the performance.
(ii) During the crucial atmospheric flight phase, the primary criterion is to reduce the load on the vehicle while minimizing the trajectory deviation, which has impact on the mission performance.
(iii) Stage transition is another major critical event wherein different conflicting requirements such as clean separation, controllability of the vehicle, smooth control transition, propulsion systems transition and performance impact are to be analysed. Based on the analysis optimum sequencing events are to be arrived at.
(iv) While separating the spent stage, it is essential to ensure that the stage impact is in the safe zone.
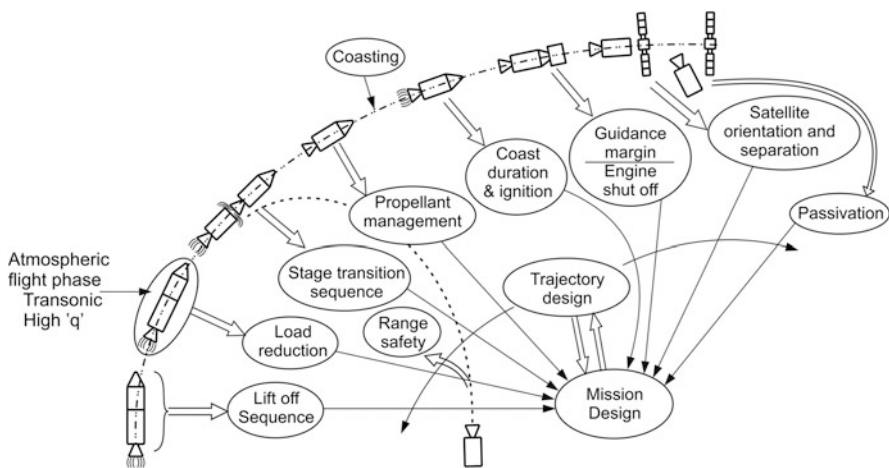(v)   Propellant management in liquid stages has to ensure engine safety while maximizing the mission performance.

Fig. 7.1 Mission design process

(vi) Coast duration (for the cases of mission with long coast between two propulsive stages) having a major impact on the vehicle performance has to be carefully designed to meet the requirements.

(vii) It is essential to ensure sufficient velocity reserve (guidance margin) in the command cut off stages while minimizing the performance loss.

(viii) Optimum mission design has to ensure minimum propellant consumption.

(ix) The mission has to meet the satellite requirement of injecting into orbit with desirable orientation. Mission sequencing has to ensure sufficient separation between the satellite and spent stages and among satellites for the cases of multiple satellite missions.

(x) Once the satellite is injected, it is essential to passivate the spent stage propulsion system to avoid collision with satellites in case of inadvertent re-ignition.

(xi) During the entire mission, it has to be ensured that the thermal environment to the vehicle sensitive systems and satellite be within the allowable limits.

(xii) It has to be ensured that commanded vehicle rates are within the allowable capabilities of control systems and the realized rates are within the vehicle structural limits.

(xiii) During the entire mission, the radio visibility of the vehicle from the appropriate ground stations is important to ensure (a) acquiring the telemetry data and (b) sending telecommand to the vehicle to destruct in case of malfunction.

All the above requirements have to be considered in the vehicle trajectory design and optimum trajectory has to be determined to maximize the performance while satisfying the constraints and requirements in (i) to (xiii).

7.3   Mission Design Strategy

The interactions among the various disciplines of STS and their interfaces with mission design process are broadly represented in Fig. 7.2.

The mission requirements and mission constraints are the major functional requirements for the mission design. The mission requirements are to be reflected as the STS performance requirements for the mission design. The performance requirement in turn decides the propulsion system design and propellant loadings. The performance reserve to be built into the system has also to be decided, and all these parameters are determined through the process of the mission design and analysis.

Similarly, the vehicle load acting on the vehicle depends on the flight environment, vehicle aerodynamic characteristics, the thrust–time curve shapes and the vehicle trajectory parameters. Trajectory analysis and thrust–time curve shaping are essential to contain the vehicle loads within the specified limits. This in turn affects the vehicle performance and has major impact on the vehicle propulsion system
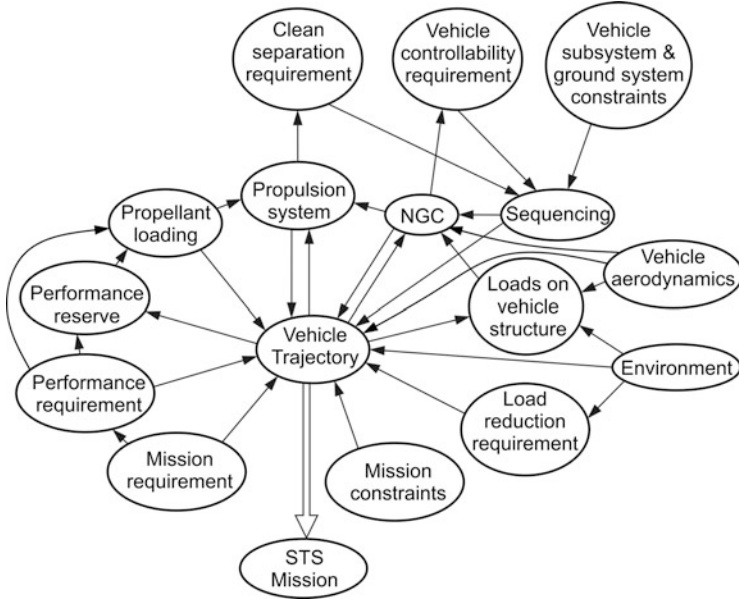
Fig. 7.2 Interactions among vehicle subsystems and mission design

parameter definition. Thus there is strong coupling between vehicle loads, propulsion system, vehicle trajectory and mission performance.

Another typical area of interaction during mission design is on the vehicle sequencing during stage transition. In this phase, almost all the systems requirements are conflicting in nature, and one has to arrive at an optimum sequence which meets all the requirements to ensure mission success. The stage transition has to be planned at low or near-zero thrust for clean separation, but it is not advantageous to meet the expected mission performance and vehicle controllability. If the additional systems are introduced to meet the requirements of separation and controllability, the vehicle becomes complex and reliability reduces. In addition, there is heavy performance loss with such additional systems. To achieve optimum sequencing, it is essential to decide event sequencing during the flight, based on real-time performance of the vehicle systems as per the specified performance criteria. This has to be implemented in navigation, guidance and control (NGC) system, and hence it brings in a strong coupling between NGC, propulsion and mission design.

During mission design, it is essential to ensure that the vehicle command rates are within the capability of control power plants and the realized vehicle rates are limited within the maximum allowable vehicle loads. To reduce the vehicle loads as per design, the real execution of the load reduction during the flight is implemented by the vehicle autopilot. This process causes a strong interaction between vehicle autopilot, control power plant, vehicle structure and mission design.

The optimum trajectory generated through the mission design is used as reference and the trajectory parameters so derived as the major inputs for the navigation, guidance and control system design. The finalized design is tuned to meet various vehicle and subsystem requirements and has impact on the vehicle performance. While designing the vehicle trajectory, it is essential to ensure that the specified range safety constraints are met. The range safety constraints during a specified stage separation also adversely affect the vehicle performance.

Thus it can be seen that the mission design has a strong coupling with almost all the disciplines of the vehicle systems and demands integrated systems design approach. To achieve the optimum design of the vehicle, the mission design has to start right at the beginning from configuration definition phase and undergo several iterations till the STS mission is firmed up.

Initially, the vehicle is configured with propulsion systems to achieve a specified equivalent velocity. Once the external vehicle configuration is finalized, with preliminary aerodynamic data, it is essential to carry out the preliminary mission design to determine the vehicle performance considering the velocity losses due to gravity and drag, pertaining to the specified mission, satisfying the entire vehicle and mission-related constraints. If the required performance is not achieved, the propellant loading or vehicle-related constraints and subsystem specification may have to be revised to meet the requirements. In this process, the requirements of performance parameters and subsystem limits are finalized.

With the finalized system parameters first-level design of subsystems is carried out along with the aerodynamic characterization of the vehicle through detailed wind tunnel tests. This process has to be repeated at various levels of development, incorporating more and more refined design parameters which are derived either through detailed analysis or experimentation. The final mission design is to be verified using all the pre-flight vehicle data.

7.4   Mission Specification Requirements and Constraints

The STS mission design has to start with a clear definition of mission objectives defining the broad goals, the mission has to accomplish. The design also has to cater for different types of spacecraft injection requirements which in turn depend on satellite applications such as remote sensing, communications, scientific investigations, interplanetary exploration, navigation, etc. When a single orbital mission is configured to cater to different satellites, it becomes necessary to define multiple objectives. Similarly if multiple satellites are to be injected in different orbits in a single STS mission then it is required to define the multiple objectives. The baseline objective in all such missions is to inject a satellite of a specified mass into a specified orbit defined in terms of orbital elements.

7.4.1   *Mission Specifications*

STS mission specifications are derived from two sources: (1) satellite requirements and (2) specified launch site and the associated constraints.

The satellite requirements are specified in terms of payload mass, payload envelope, orbital requirements and injection accuracy, satellite attitude requirements and the limits on attitude rates at the time of injection. The specified envelope of the satellite is the requirement for deciding the payload fairing (PLF) configuration acceptability or for the reconfiguration of PLF if the need arises. This forms part of the vehicle configuration.

It is to be noted that even though a nominal satellite mass is defined, in reality the specified mass cannot be realized exactly. Therefore the possible dispersion on the realized satellite mass has to be specified to meet the defined orbital requirements.

As per the satellite application requirements, nominal values of orbital elements are specified which need to be achieved by the STS. But in reality, the exact values of injection orbital elements are not possible due to the presence of dispersions of the engine shut-off characteristics, errors of navigation sensors and the errors caused by the guidance and control algorithms in the real operating environment. The propulsion system of the satellite has to correct the orbital errors caused by the STS mission. Depending on the capability of satellites to correct such errors, allowable dispersions of the orbital elements have to be specified. The STS systems have to be designed to meet the mission requirements as defined, within the specified dispersion band. During initial phase of vehicle design process, a trade-off study has to be carried out between the satellite capability and the STS system requirements. An optimum set of dispersions agreeable to both the systems have to be arrived at based on the preliminary mission design, and the corresponding specifications for the STS mission are to be generated.

The satellite demand is to achieve all the six orbital parameters simultaneously. Generally, two of the orbital parameters, viz. the longitude of ascending node (Ω) and true anomaly (θ), can be achieved by suitably selecting the day and time of launch from the specified launch site, considering the trajectory duration from lift-off to satellite injection.

Also, it is to be noted that generally the satellite orbital requirements are in terms of mean orbital elements, whereas in reality the STS mission achieves the osculating orbital elements corresponding to the locations of satellite injection. Therefore, considering the satellite requirements, STS trajectory and injection location, the osculating elements at the injection of satellite are to be worked out and defined as the specifications for the STS mission.

The specifications for STS mission defined by launch site are decided based on the suitability of identified launch site and limits on the allowable launch azimuth. The launch site location is defined in terms of longitude and geodetic latitude. From the specified launch site, based on the allowable launch azimuth corridor, the most suitable planar trajectory providing maximum performance has to be specified. If the optimum azimuth is beyond the allowable corridor, then the required orbital

inclination is not feasible to achieve. In such cases, there are two ways of managing the mission: (1) Maximum possible launch azimuth which provides orbital inclination close to the satellite requirement is specified. The exact inclination requirement of the orbit can be further achieved by the satellite systems. For such cases, the maximum possible launch azimuth and the corresponding orbital inclinations are the specifications for STS mission. (2) Initial flight of STS is along the launch plane specified by the maximum possible launch azimuth. Once the specified range safety boundaries are crossed, the vehicle has to follow three-dimensional trajectory motions, to achieve the required inclination. To achieve the maximum performance in three-dimensional trajectory, the yaw manoeuvre has to be initiated at the early part of the trajectory. But it has to be initiated only after meeting (a) the range safety constraints and (b) load constraints of the vehicle. In all such cases, the maximum possible launch azimuth, the required orbital inclination and the time of initiation of yaw manoeuvre are the specifications for the STS mission.

The launch tower at a given launch site is invariably configured in the specified direction as per the requirements of ground systems and ease of vehicle integration. In such cases the vehicle can be assembled at the launch tower only in a specified orientation depending on the various fluid lines, umbilical interfaces between the ground systems and vehicle. The vehicle pitch plane at launch pad may not align with the launch plane defined by launch azimuth. Under such conditions, the navigation system in the vehicle at the time of lift-off senses the difference in the vehicle roll orientation with respect to the launch azimuth plane. This difference is equal to the bias of launch tower orientation with respect to the launch direction. To correct this difference the vehicle is commanded to roll once the vehicle crosses the launch tower vertically. The roll angle is decided such that the vehicle pitch plane is aligned with the launch plane (defined by launch azimuth direction). The roll angle, time of initiation of roll manoeuvre and commanded roll rate also become the mission specification for STS.

Considering the above aspects, typical STS mission specifications are summarized as given below:

(a) Satellite mass: $m_{nominal} \pm \Delta m$
(b) Osculating orbital elements at injection:

    1. In the case of circular orbit,

       Orbital altitude: $h_c \pm \Delta h_c$

    2. In the case of elliptical orbit,

       Perigee altitude: $h_p \pm \Delta h_p$
       Apogee altitude: $h_a \pm \Delta h_a$
       (or)
       Semimajor axis: $a \pm \Delta a$
       Eccentricity: $e \pm \Delta e$

    3. Orbital inclination: $i \pm \Delta i$
    4. Argument of perigee: $\omega \pm \Delta\omega$

(c)   Satellite orientation at the time of injection:

Local pitch angle: $\theta_L \pm \Delta\theta$
Local yaw angle: $(\psi_L \pm \Delta\psi)$
Local roll angle: $\phi_L \pm \Delta\phi$

(d)   Attitude rate at the time of satellite injection:

Pitch rate: $\leq q_l$
Yaw rate: $\leq r_l$
Roll rate: $\leq p_t$

(e)   Launch station coordinates:

Geodetic latitude: Ø_{GDL}
Longitude: λ_L

(f)   Launch azimuth: A_{ZL}
(g)   Time of launch: **H** hrs : M min : S seconds
      The launch time can be specified in terms of universal time or can be converted into local time at the time of lift-off.
(h)   Time of initiation of yaw manoeuvre: t_Y
 (i) Roll manoeuvre during lift-off:

Time of initiation of roll manoeuvre: t_i
Time of stopping roll manoeuvre: t_e
Command roll rate of vehicle: p_{cl}

The STS mission design has to be carried out with the above specification. The system requirements and constraints to achieve the above mission by STS are explained in the following sections.

7.4.2   Mission Requirements

Once the broad mission specifications are defined, the next logical step is to specify the requirements and constraints and carry out requirements analysis. This analysis has to lead to matching the satellite and launch vehicle capability with the defined mission goal. The vehicle should be able to deliver the required velocity to the satellite, taking into account all losses during the flight and various constraints to guarantee the desired orbit. Important requirements are as given below:

(a)   Defining a suitable vehicle configuration capable of meeting the mission goal.
(b)   Identifying the suitable vehicle subsystems to meet the defined mission within the specified dispersion band.
(c)   Generating an optimum suitable vehicle trajectory meeting all defined constraints.

(d) Ensuring the vehicle visibility throughout the entire phase of flight and clearly defining the visibility constraints if any.

(e) Specifying the maximum dynamic pressure and loads on the vehicle to ensure that the vehicle structure experiences the loads which are well within the maximum specified loads with adequate margin.

(f) Shaping thrust profile to reduce the dynamic pressure while meeting the required mission performance. The finalized thrust profile is the requirement for the propulsion system.

(g) Minimizing the loads if necessary by selecting a suitable strategy for load relief.

(h) Defining range safety requirements to ensure safe impact of the spent stages.

(i) Finalizing the propellant loading considering the mixture ratio dispersions to ensure maximum performance while ensuring safety of propulsion system and other subsystems.

(j) Generating the 'guidance margin' to guarantee that the spacecraft is injected into the specified 'injection pillbox', even when the propulsion systems under perform within permissible limits.

(k) Meeting all defined thermal constraints throughout the entire regime of vehicle flight.

These form major design guidelines/considerations/constraints in trajectory design and other mission studies. If the vehicle for a given mission is finalized, major changes in vehicle configuration may not be possible at this stage. However, a few changes like propellant loading in the propulsion stages, etc. meeting the overall constraints may be attempted.

7.4.3 Mission and Vehicle Constraints

While defining the mission, a clear understanding of all constraints stemming from the vehicle environment, vehicle safandety, range safety and all other associated areas is needed. Some of the important constraints are

(a) Axial acceleration limits on lift-off and stage separation, the acceleration levels are to be maintained within the maximum level with respect to structural loads as well as on humans in case of human space missions, etc.

(b) Flight safety considerations to ensure that the impacts of spent stages are only in international water

(c) Identifying the non-visibility zones for appropriate action

(d) Vehicle loads during the atmospheric region, considering vehicle environmental factors

(e) Maximum vehicle attitude rates and angular acceleration as specified by vehicle subsystems

(f) Critical thermal constraints on vehicle and spacecraft systems

These have to be considered while carrying out detailed mission studies.

7.4.4   Mission Studies

The various studies needed for the launch vehicle mission encompass several areas. The following studies which are highly interactive form the basis to firm up the overall mission:

(a)   Configuration finalization to meet the spacecraft requirement
(b)   Aerodynamics design and analysis
(c)   Adequacy of propulsion
(d)   Design of a suitable optimum trajectory
(e)   Control and guidance design
 (f)   Mission sequencing
(g)   Evaluation and definition of flight environment
(h)   Performance analysis considering dispersions to assess the effect of variation of different system parameters on mission

All these studies demand accurate generation of vehicle data, which includes data in terms of mass details, mass properties like centre of gravity, mass moment of inertia, aero-propulsion, actuators, sensors, flexible characteristics, slosh, etc. The vehicle sign convention has to be clearly spelled out.

The overall activities required for realizing a successful mission are detailed in Fig. 7.3. These activities play a vital role in a launch vehicle mission, right from the conceptual design. Various studies needed in respect of (a) configuration finalization, (b) aerodynamics design and analysis, (c) adequacy of propulsion, (d) aerothermal design and analysis, (e) control and guidance design and (f) definition of flight environment have been described in detail in appropriate chapters of this book.

7.5   Trajectory Phases and Mission Design Tasks

Broad trajectory phases of a typical STS mission are represented in Fig. 7.4. The vehicle trajectory during ascent phase can be divided into two distinct phases, the atmospheric phase and the exo-atmospheric phase. The initial portion of ascent trajectory where the vehicle negotiates through the dense regions of the atmosphere is a critical phase of flight with the wind playing a significant role on the launch vehicle design process. In the exo-atmospheric phase generally the stage impact, separation of stages and payload fairings and visibility from tracking stations are of importance.

The atmospheric phase of flight can be further divided into three segments as shown in Fig. 7.4. Initial vertical flight of the vehicle is known as lift-off phase, when the vehicle moves vertically to several tens of meters to clear the launch pad. This is essential to avoid the collision of vehicle to the launch tower against various disturbances acting on the vehicle. The next segment refers to the initial pitch-down
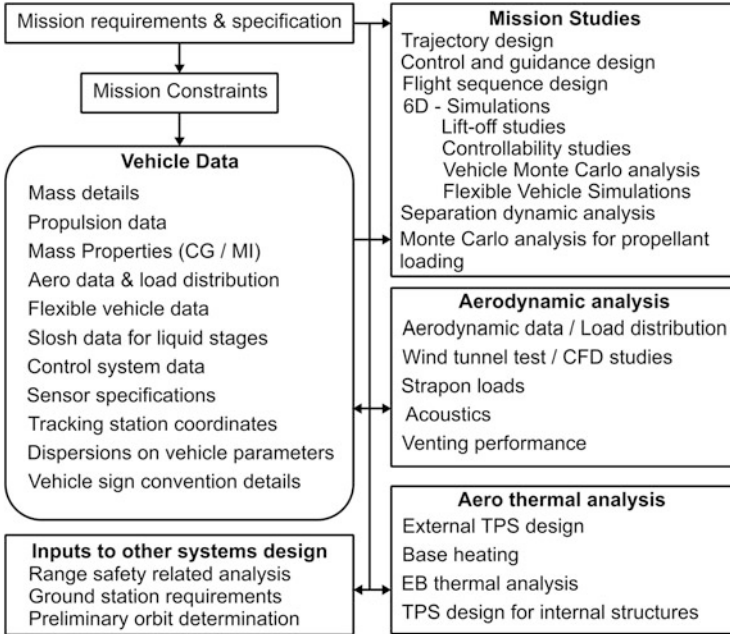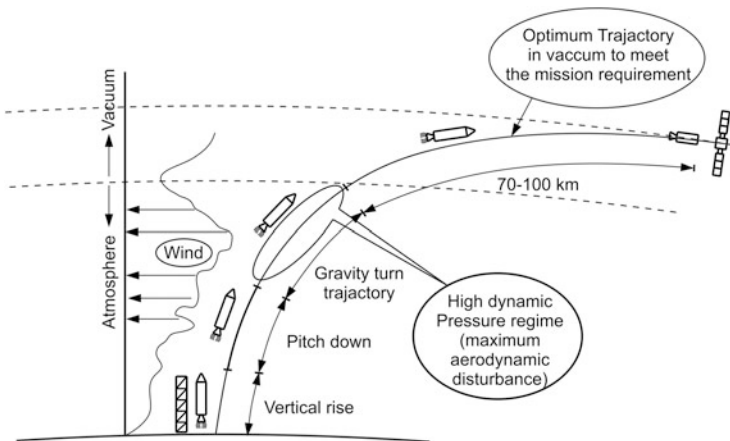
Fig. 7.3 Overall mission activities

Fig. 7.4 Typical trajectory phases of an STS mission

phase after the vertical flight where it is possible to introduce larger manoeuvres since the dynamic pressure is low due to lower vehicle velocity. The higher angle of attack caused by large manoeuvre at this phase does not cause large loads on structures.

Third phase of atmospheric flight extends up to an altitude of about 40–70 km where atmospheric wind effects are significant. Generally the vehicle is steered along gravity turn trajectory in this segment to keep the angle of attack to a very low value and thus achieving minimum structural load on the vehicle. Zero angle of attack during this phase means minimum thrust vectoring of engines to keep the vehicle along gravity turn trajectory, and this leads to utilizing the maximum thrust of the motor for vehicle acceleration. The control in this regime is active and keeps the vehicle stabilized against disturbances. However spending more time in gravity turn trajectory during atmospheric flight region is an undesirable feature. In most of the vehicles, till the vehicle attains an altitude of about 70 km, open-loop steering programme is used for guiding the vehicle.

The exo-atmospheric phase as shown in Fig. 7.4 is beyond 70–100 km depending on vehicle characteristics and mission constraints till satellite injection. This phase allows controlled manoeuvre, and higher attitude rate commands from guidance do not cause any structural problem to the vehicle. Therefore closed-loop guidance algorithms are used in this phase, but the angular rate and acceleration of the vehicle are limited to the acceptable values from overall mission considerations. Higher angular rate to vehicle is affected using the higher thrust vectoring of the vehicle. Since this type of manoeuvre causes higher steering loss in the total vehicle velocity, it is advantageous to have higher manoeuvre (especially yaw manoeuvres) as far as possible during the earlier phases of vehicle where the velocity is lower.

Mission design tasks to be carried out during various phases of STS mission can be broadly categorized as given below:

(a) Lift-off studies
(b) Load relief methodologies
(c) Thermal loads on the vehicle
(d) Mission sequence design
(e) Propellant loading requirements
(f) Velocity reserve requirements
(g) Satellite injection orientation
(h) Propulsion stage passivation requirements
(i) Tracking and visibility requirements
(j) Range safety issues

Considering the various requirements and constraints of the above mission activities, an integrated trajectory design and analysis has to be carried out to maximize the vehicle performance to the specified satellite mission.

The details of these tasks and their design considerations are explained in the following sections.

7.6 Lift-Off Studies and Analysis

Lift-off phase is one of the crucial phases of a STS mission. Generally, the first stage ignition for the vehicle is commanded by the ground checkout computer system. All the further sequencing commands are issued by vehicle on-board computers. Therefore, to ensure safety of the mission, vehicle, ground systems and facilities, it is essential to confirm that the first stage performance is as per the expectations. This can be achieved through two-tier logic as represented in Fig. 7.5. At a specified time from the ground ignition command, performance of the engine is checked by verifying the suitable engine performance parameters. Once the performance is within the specified bounds, further activities are allowed to proceed. In case the engine performance is outside the specified bounds, then shut-off command is issued and mission is called off. If the performance check is passed, then the physical lift-off of the vehicle is checked. This is done through confirming the connector demating status of last-minute plug or vehicle sit-on umbilical connector. If the vehicle physical lift-off is confirmed, then only vehicle on-board sequencing is commenced. All the further on-board sequencing is referred with respect to the physical lift-off time. Otherwise mission abort sequence is activated for saving the vehicle and calling off the launch.

The main design parameters during lift-off phase of STS mission are

1. On-board lift-off sequence (ignition time, control initiation time, etc.)
2. Vertical flight time, after which the pitch-down manoeuvre starts
3. Performance parameters, nominal values, their bounds and time of initiation for performance confirmation
4. Physical movement criteria and the corresponding values
5. Time of initiation and end of intentional roll manoeuvre
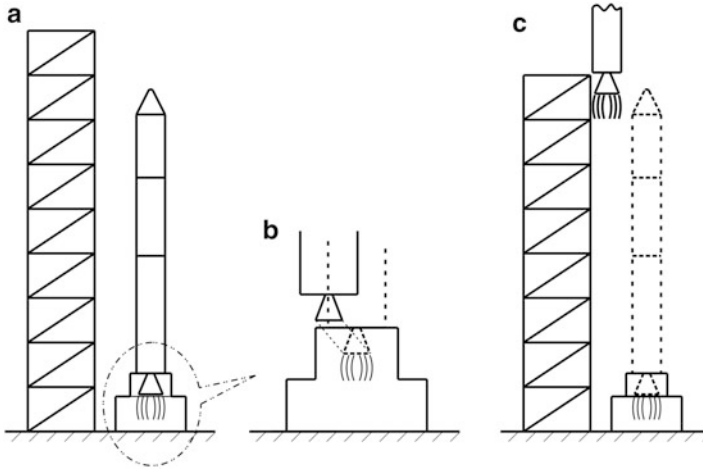

Fig. 7.5 Lift-off clearance strategy
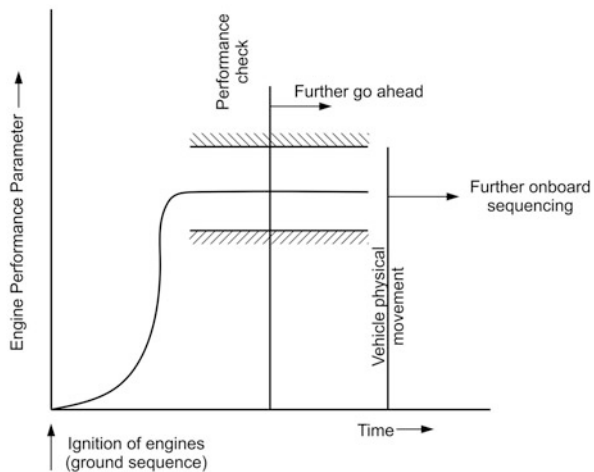
Fig. 7.6 Lateral movement during lift-off phase (**a**) Vehicle on pad (**b**) Lift-off (**c**) Clearing tower

The above design parameters have to be arrived at to meet the following requirements:

1. Clean lift-off of the vehicle with respect to the launch pedestal
2. Minimize the lateral drift of the vehicle towards launch tower
3. Reduce the thermal loads on the launch pad and launch tower
4. Minimize the stay-off of the vehicle over launch pad to reduce the jet acoustic loads and thermal loads
5. Vertical rise time has to be optimized to achieve maximum performance of the vehicle.

The various disturbances which affect the lift-off clearance are thrust misalignments of booster motors, differential thrust for strap-on motors (if any), lateral thrust offset, navigation system pointing error, pull-out forces of the umbilical cord, aerodynamic disturbances and surface winds. Since the direction of surface winds keeps on changing, appropriate direction which can cause the movement of vehicle towards tower is to be considered. These disturbances make the vehicle move laterally as shown in Fig. 7.6. Any collision during lift-off phase leads to the mission catastrophe. By suitably selecting the physical lift-off criteria, along with control design and control initiation time, the lateral displacements due to disturbances can be reduced to a greater extent. Therefore, suitable design of these parameters has to be carried out to ensure the safe clearance of the vehicle from umbilical tower even under worst-case disturbances. The safe clearance needed is to be checked not only with the launch tower but between the nozzles and launch pedestal, as shown in Fig. 7.6, and interference of the vehicle fin (if provided with) with any projection of launch tower during the ascent phase.

The vertical rise time along with the (T/W) ratio severely affects the mission performance of the vehicle. This time decides the vehicle stay time over the launch pad, which has major impact on jet acoustic loads on the vehicle as well as thermal loads on the launch pedestal. Initiation of pitch-down along with the vehicle lateral drift has an impact on thermal environment to the launch tower due to the interaction of jet exhaust of the engine with the tower. Therefore, the height of the vertical rise has to be judiciously selected considering all the above aspects.

7.7   Vehicle Load Reduction Strategies

The atmospheric phase flight of the vehicle after the initial pitch-down is quite complex. In this phase, as the dynamic pressure keeps increasing, the aerodynamic forces become significant. Therefore, the velocity loss due to aerodynamic drag increase is significant, and the increased lateral aerodynamic force associated with the increased control demand has the tendency to increase the loads beyond acceptable limit and break the vehicle. Not only the various losses due to drag, gravity and steering are to be minimized during the ascent phase but also the structural loads, to avoid the collapse of structure due to excessive loads beyond the design limits. The maximum load on the structure during the atmospheric stage is due to aerodynamic load which is characterized by the product of dynamic pressure Q and angle of attack α, that is, Q α, as explained in the different chapters of this book.

The important parameters and features of atmospheric flight phase are represented in Fig. 7.7. As the vehicle rises through the atmosphere, the velocity and altitude of the vehicle increase continuously, whereas the density of the
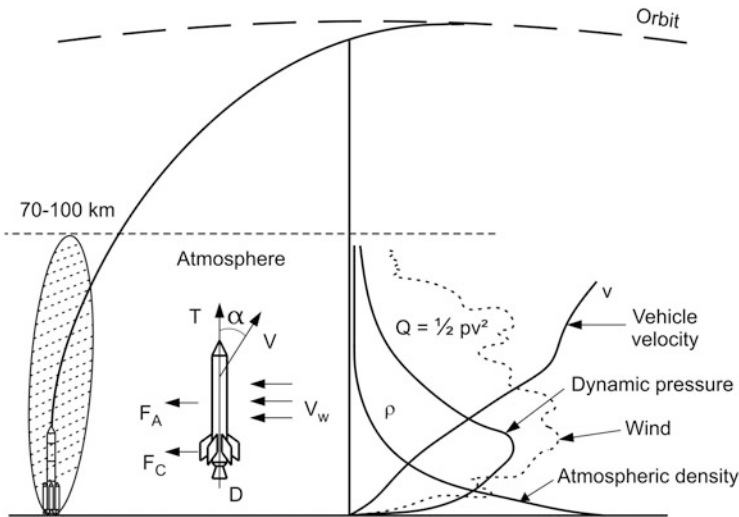

Fig. 7.7   Criticalities of atmospheric flight phase

atmosphere which is a function of altitude decreases. As the vehicle velocity increases, the dynamic pressure, $Q = (\frac{1}{2})\rho V^2$, also increases and the peak value of Q occurs when the vehicle reaches somewhat higher velocity where the density is still significant. Generally, the peak dynamic pressure occurs during the altitude range of 7 to 15 km. Atmospheric wind also generally peaks during this altitude regime. Wind velocities are the major contributing factor for inducing angle of attack α to the vehicle. Since the aerodynamic disturbance forces and vehicle loads are functions of Qα, the combination of higher dynamic pressure and higher wind velocity occurs almost simultaneously creating a complex and high disturbing environment to the vehicle systems.

During the atmospheric phase of flight, the wind too plays a major role on dynamics of the launch vehicle. These factors induce aerodynamic force and moments as well as aerodynamic load on the vehicle. Therefore a proper strategy which minimizes the aerodynamic loads on the vehicle is required. The aerodynamic load in this region has to be minimized either by reducing Q or by restricting the maximum α or both.

7.7.1   Thrust Profile Shaping

The dynamic pressure profile depends on the velocity build-up with respect to the vehicle altitude. The velocity profile depends on thrust profile of the booster motors. Therefore, the dynamic pressure profile during atmospheric flight phase is essentially decided by the shape of the thrust profile of the propulsion system. To reduce the dynamic pressure, the thrust profile of the vehicle has to be such that the vehicle rises quickly to the higher altitude when the vehicle velocity is lower and the major velocity build-up happens at higher altitude. This demands higher thrust initially, and the thrust values have to be lower during the critical regime of flight, till the vehicle reaches sufficient altitude. Subsequently, the thrust has to be increased to build the velocity as per the mission requirement.

For liquid engines, the thrust value is generally constant. By throttling the engines, the required thrust profile as represented in Fig. 7.8a can be generated to achieve the reduced dynamic profile. In solid motors, generally, progressive thrust–time profiles are used. To meet the requirements of reducing dynamic pressure, the propellant grains of different segments of the motors can be suitably designed to generate the required profile as represented in Fig. 7.8b.

The thrust shape of booster motors in lower stages has impact on the vehicle and propulsion system performance. Therefore, while shaping the thrust profile to reduce the dynamic pressure, it is essential to ensure that the profile achieves the required performance and the shape of the profile is within the capability and constraints of the selected propulsion system. The derived thrust profile forms a major input for the propulsion system design.
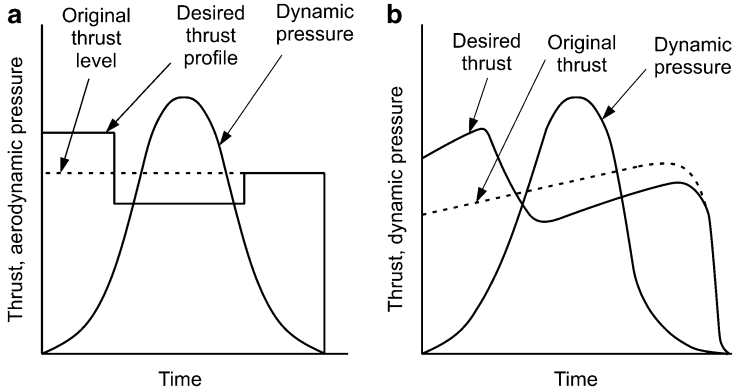
Fig. 7.8 Booster phase thrust shaping (**a**) Liquid engines (**b**) Solid motors

The aerodynamic load on the vehicle with the maximum possible reduced dynamic pressure is still beyond the vehicle capability; further strategies are to be considered in design to reduce the angle of attack.

7.7.2 Load Relief Systems

The major contributing factor for creating angle of attack is atmospheric winds. Hence it is necessary that efforts are to be made to reduce wind-induced angle of attack, α, during the atmospheric phase of flight. Such methods are called load relief systems. Basically the load relief system steers the vehicle into the wind, thus reducing the angle of attack. Load relief has to be attempted mainly during the high–dynamic pressure region of the flight, and duration of the load relief is to be critically analyzed based on the vehicle and trajectory. It may be noted that if the load relief trajectory is not attempted, launch availability in certain seasons gets reduced due to increase in the angle of attack beyond the allowable values.

The load relief can be done either by (a) using lateral accelerometer feedback (active load relief) or (b) designing the wind bias steering (passive load relief system). In active load relief it is possible to use angle of attack sensor, but proper implementation of this scheme is quite complex. The lateral acceleration signal represents the angle of attack. As this system is more robust, generally, lateral acceleration signals are used for active load relief system.

In active load relief system, the lateral accelerometer package with two accelerometers, one along pitch and another along yaw axis, placed at a convenient location from the centre of gravity of the vehicle is recommended to provide load relief. This sensor provides one more parameter (in addition to attitude and body rate) in the feedback signals for attitude control. The output of lateral accelerometer

has an analogy for angle of attack sensor, and the effect of adding this signal into control command is to tilt the vehicle into wind. This helps in reducing the vehicle angle of attack and also the corresponding engine gimbal angle. The resulting effect is the reduction of bending moment on the vehicle. In such active load relief control, it is possible to select suitable gains in the feedback signals to achieve the drift minimum or load minimum control depending on the requirements.

In drift minimum system the aerodynamic and control forces are fully balanced, and hence the drift of the vehicle is minimized. But it provides load relief only to some extent. Alternatively if the weightage to the attitude feedback is reduced the focus would be on minimizing the angle of attack which in turn causes considerable reduction of loads on the vehicle. However this causes the vehicle to drift away from the desired trajectory, which has to be corrected by the guidance system subsequently. Therefore it is essential to have a design which offers the best combination of load minimum and drift minimum. One of the solutions is to apply the load relief only for shorter durations in regimes where the dynamic pressure is high, and the time period has to be chosen such that it results in the reduction of high aerodynamic loading. Details of active load relief system are described in Chap. 14.

Another commonly used load relief system is following gravity turn trajectory with wind biasing, which is a passive system. In this system, the steering programme is designed in such a way that it compensates for the prevailing wind conditions during launch.

7.8 Gravity Turn Trajectories and Wind Biasing

The requirement of reducing angle of attack is achieved by following the gravity turn trajectory. This manoeuvre is an important feature of atmospheric phase trajectory, where the vehicle axis is aligned with velocity vector profile continuously. Such manueuvre ensures zero angle of attack thereby reducing the vehicle load to the barest minimum.

7.8.1 Gravity Turn Trajectories

In gravity turn trajectory manoeuvre, the vehicle axis (thrust direction) is aligned with the velocity vector. Consider STS as an axi-symmetric body and thrust direction along the longitudinal axis of the vehicle. Vehicle orientation and gravity turn trajectory of such a vehicle is represented in Fig. 7.9. For the gravity turn trajectories of axi-symmetric vehicles, the aerodynamic drag force acts on the vehicle whereas the lateral force is zero due to zero angle of attack.

Fig. 7.9 Gravity turn trajectory
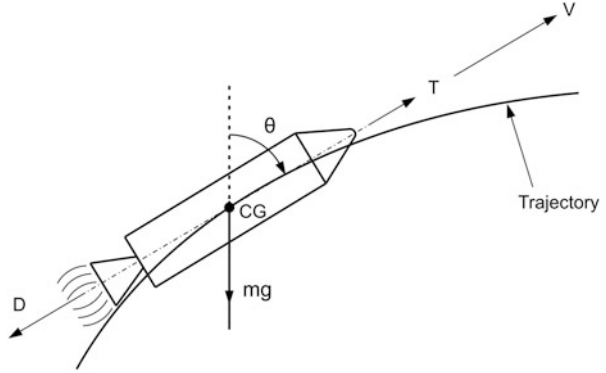
Thus, the equations of motion of the vehicle following gravity turn trajectory represented along the vehicle longitudinal axis and normal to the vehicle axis are given as

$$m\frac{dV}{dt} = T - D - mg\cos\theta \tag{7.1}$$

$$mV\frac{d\theta}{dt} = mg\sin\theta \tag{7.2}$$

where m is the vehicle mass, V is the vehicle velocity, T is the thrust force, D is the aerodynamic drag force, g is the acceleration due to gravity at the flight instant and θ is the vehicle attitude with respect to the local vertical. From Eq. (7.2),

$$\dot{\theta} = \frac{d\theta}{dt} = \frac{g\sin\theta}{V} \tag{7.3}$$

The vehicle attitude rate to follow the gravity turn trajectory is given by Eq. (7.3). It can be seen from Eq. (7.3) that if there is no gravity, the vehicle trajectory is a straight line with constant vehicle attitude. The presence of gravity makes the trajectory to have a shape of curvature, and hence this trajectory is known as gravity turn trajectory.

It is to be noted that the vehicle attitude rate depends on vehicle instantaneous attitude, velocity and gravity to follow the gravity turn trajectory as given in Eq. (7.3). The following are the important aspects of gravity turn trajectories:

1. To have a meaningful gravity turn trajectory, vehicle has to achieve a certain velocity. Otherwise the required vehicle rates are very high.
2. At the end of vertical rise, the vehicle attitude with respect to vertical is zero. Therefore, the gravity turn trajectory initiation at the end of vertical rise is not possible.

Fig. 7.10 Gravity turn rate
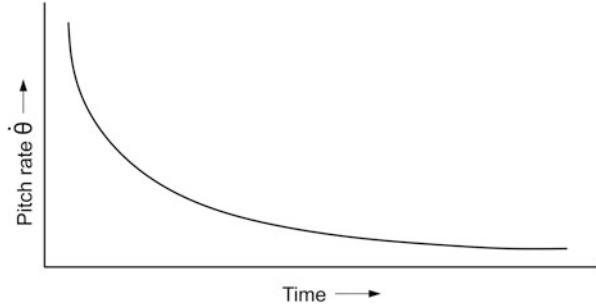
3. Combining these two features, to achieve a realistic gravity turn trajectory, the gravity turn has to be initiated after intentional pitch-down manoeuvre of the vehicle.

Since the vehicle velocity is lower, the gravity turn rate can be relatively higher at the initial phase and subsequently can be reduced depending on the velocity build-up and instantaneous attitude of the vehicle as shown in Fig. 7.10.

It is to be noted that if the gravity turn is initiated at much early stage when the vehicle attitude is near vertical then the gravity turn rate is small and leads to steeper vehicle trajectory in atmospheric flight phase, ending up with more velocity loss due to gravity. Thus the mission performance of the vehicle drastically reduces. For the cases with larger θ, early initiation of gravity turn demands higher vehicle rates for the vehicle. These aspects demand that the gravity turn be initiated as late as possible. But to reduce the vehicle load, the gravity turn has to be initiated as early as possible. Also, at the initiation of gravity turn, there may be discontinuity between actual vehicle rate and the gravity turn rate demand at that instant.

Therefore, vertical lift-off time, pitch-down manoeuvre duration and transition phase as represented in Fig. 7.11 are decided judiciously to meet all the requirements. The intentional pitch-down manoeuvre rate after vertical rise is decided to meet the performance requirements at the end and to have smooth transition to gravity turn manoeuvre. The timings for these events are designed to meet the performance and load requirements simultaneously.

7.8.2   Wind Biasing

During the STS flight with gravity turn trajectory, if there is no wind, velocity vector is always aligned to the longitudinal axis of the vehicle as represented in '1' of Fig. 7.12. But the prevailing wind V_W at higher altitude makes the relative velocity of the vehicle away from the longitudinal axis, as represented by V_R in the figure. Thus, the vehicle flight direction is along V_R, which makes the angle of attack α to the vehicle axis as shown in Fig. 7.12

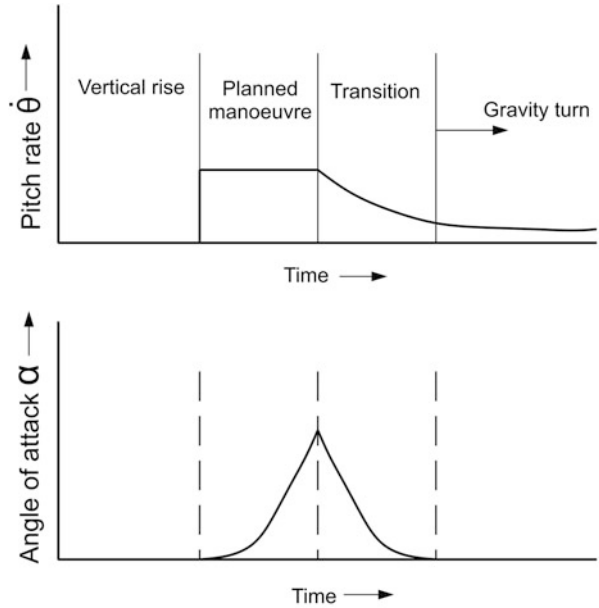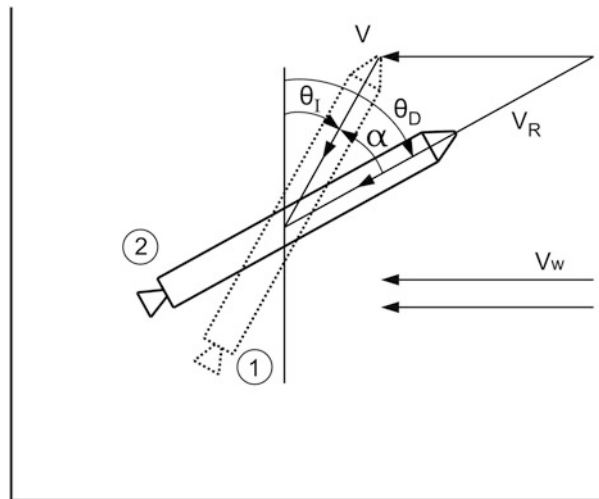**Fig. 7.11** Gravity turn
trajectory transition

Fig. 7.12 Angle of attack
and wind biasing

The angle of attack induces disturbing aerodynamic normal force on the vehicle.
This force tends to rotate the vehicle about the centre of gravity, and the control
system of the vehicle generates the necessary control force to stabilize the vehicle
against the disturbances. The combined aerodynamic normal force and the
balancing control force in turn introduce load on the vehicle structure. The aero-
dynamic load indicator '$Q\alpha$'of a typical STS during its mission through a typical
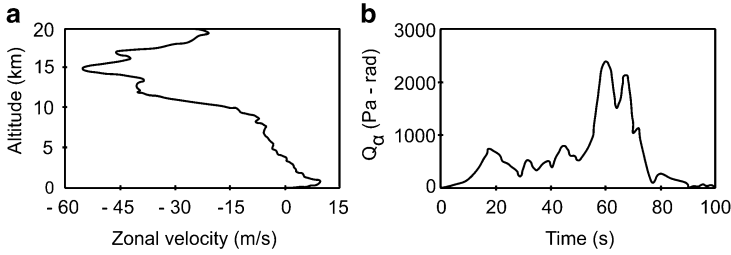
Fig. 7.13 Impact of wind on aero load indicator (**a**) Zonal wind velocity (**b**) Aero load indicator

measured wind profile is given in Fig. 7.13. It is seen that even though the trajectory is designed with zero angle of attack with no wind conditions, the in-flight wind induces considerable value of 'Qα'. To reduce Qα, it is essential to reduce the angle of attack α.

The most efficient way of reducing angle of attack is to fly the vehicle such that longitudinal axis of the vehicle is aligned with V_R, which is caused by wind velocity as shown in '2' of Fig. 7.12. This feature is called 'wind biasing'.

Thus, in wind biasing trajectory, vehicle steering programme is designed such that at any instant, the vehicle attitude is always aligned with relative velocity vector. The important aspects to be considered during wind-biased trajectory are as follows:

Even though the angle of attack is represented in simple form in Fig. 7.12, the realistic contributions are represented in Fig. 7.14:

$$\alpha = (\theta_c - \theta) + \frac{V_d}{V} + \alpha_w \tag{7.4}$$

where $(\theta_c - \theta)$ is the difference between the desired attitude of the vehicle and the actual realized vehicle attitude, V_d is the lateral drift of the vehicle normal to the vehicle axis, which is caused by the control force and residual normal aerodynamic force and α_w is the angle of attack caused by the wind velocity, V_w.

Therefore, even though α is reduced to zero with wind velocity, V_w, there can be non-zero values for $(\theta_c - \theta)$ and V_d. These factors introduce lateral trajectory drift and affect the vehicle trajectory in subsequent phases, which in turn affect the vehicle performance. It is essential to design the integrated trajectory with wind biasing to achieve the required performance. Therefore, wind biased trajectory has to be attempted only when required. The initial conditions at the time of CLG initiation achieved by such design have to be considered for CLG design. The integrated steering programmes for both atmospheric flight and the CLG phases have to be validated through extensive simulations. Real on-board systems are also to be used in simulations to verify the on-board implemented logics before they are cleared for flight.
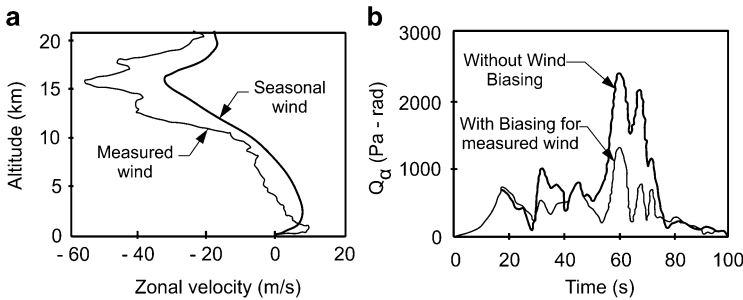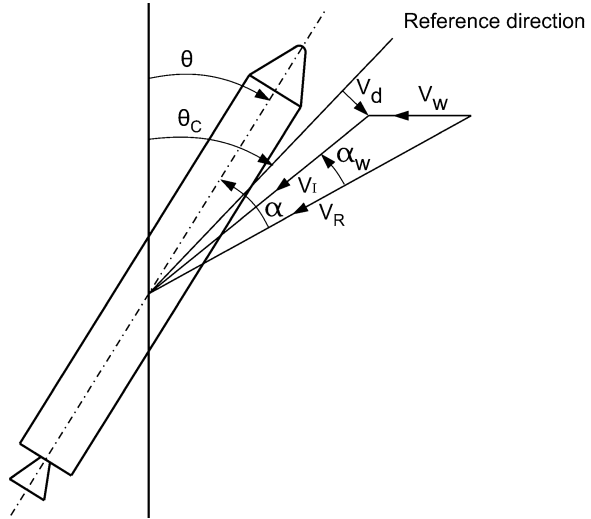
Fig. 7.14 Angle of attack
components

Fig. 7.15 Impact of seasonal wind biasing (**a**) Wind (**b**) Aero load indicator

To design wind-biased trajectory on ground, the in-flight wind profile is required. But it is to be noted that, due to the highly random nature of the wind profiles, the in-flight wind is not known a priori.

One possibility is to design the trajectory biased to the mean wind profile of the season of the launch. Due to this 'biasing', the vehicle is commanded into the mean wind at that altitude. Thus the wind-induced angle of attack is only due to difference of the actual wind prevailing at that altitude and the mean wind that is used in biasing as shown in Fig. 7.15a. Under such cases, there is considerable reduction in $Q\alpha$ as shown in Fig. 7.15b. This strategy helps in increasing the launch probability of that season.

It can be seen from Fig. 7.16 that there are large deviations in mean wind between seasons. Therefore, this design has the disadvantage if the launch is postponed to different season due to unforeseen problems. In such cases it becomes

Fig. 7.16 Seasonal mean winds
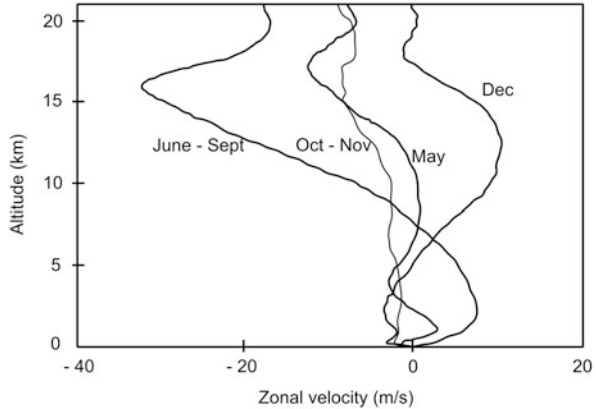
necessary to redesign the steering programme with the changed seasonal winds, which is a cumbersome process.

It is to be noted that there can be large variation in day-to-day winds measured during the launch campaign phase. Typical shape of profiles is as given in Fig. 7.17a. Therefore, the seasonal mean wind biasing still does not cater to the large variations between the seasonal mean wind and the prevailing wind during the day of launch. But the wind variations during a short period of time (within about 1 h) are very small as shown in Fig. 7.17b.

Therefore, the above drawback can be overcome by biasing the trajectory to wind that is prevailing on the day of launch. This is termed as Day-of-Launch (DOL) wind biasing. DOL wind biasing is based on the wind measurements which is as close to the launch as possible on the launch day and then generating the steering programme for this wind just before launch. The DOL wind biasing needs a reference trajectory which is capable of meeting the defined payload requirements. It is also necessary to define the target conditions at the end of open-loop trajectory phase, so that closed-loop guidance design remains unaffected. The powerful computing facilities, which are available presently, enable the process of generation of wind-biased steering programme within a short duration and its use for trajectory designs and its validation.

7.8.3 Implementation Aspects of DOL Wind Biasing

The major challenge in implementation of DOL wind biasing is to minimize the overall time needed to complete the entire process after the wind measurements are carried out very close to launch. This involves processing of measured wind data, generation of open-loop wind-biased steering programme and validation of the design in simulations. To utilize the latest wind data as close as possible to the flight time, on the launch date, the entire process of generation of trajectory has to be

Fig. 7.17 Wind
characteristics: (**a**) day-to-
day variation with respect to
mean wind (**b**) wind
variations within 1 h 20 min
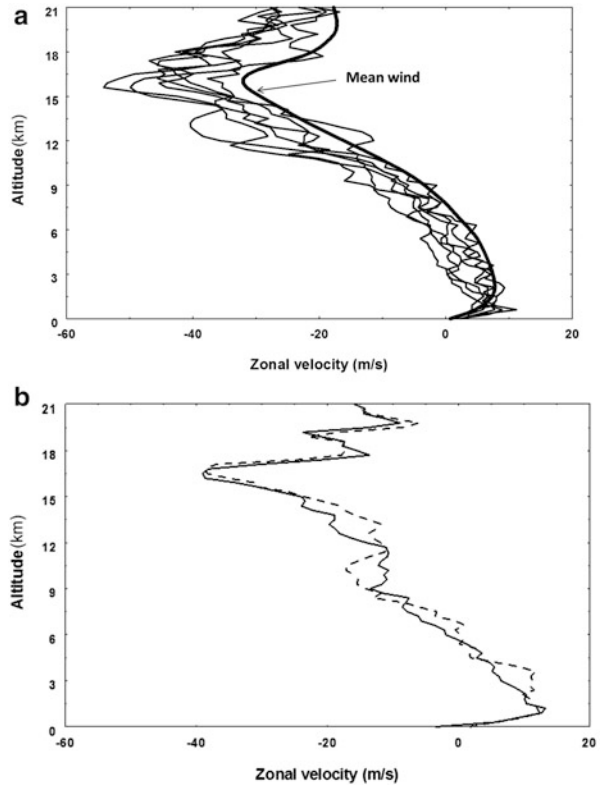
completed within the stipulated period, validated and loaded in the on-board
computer during the countdown sequence. This demands automation of all
processes.

The approach needed is that a reference open-loop trajectory for a mission and a
closed-loop guidance (CLG) suitably designed are available and validated through
extensive evaluation through several phases of simulations. Entire CLG programme
and data are to be coded and stored in flight computer. It is important to define the
initial conditions for a pillbox at the end of open-loop trajectory and to ensure that
the vehicle parameters meet these target conditions within allowable dispersions.
Therefore the generation of DOL open-loop steering programme has to ensure that
the specified end conditions are always met so that CLG design remains insensitive
to wind variations and functions smoothly. This helps to avoid the revalidation of
CLG algorithm which has been already designed and tested extensively.

Once the DOL wind-biased steering programme is generated, it is to be inte-
grated with the total vehicle trajectory including the closed-loop guidance and
validated using all digital simulation test beds and also in integrated on-board
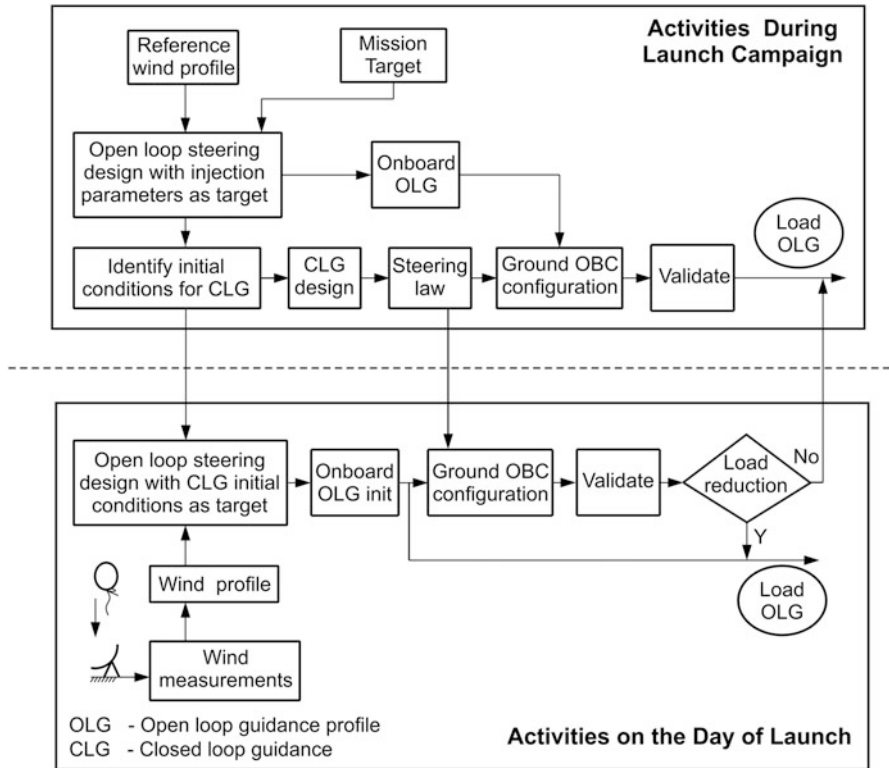
Fig. 7.18 Day-of-Launch wind biasing design and implementation strategy

equivalent systems. The major implementation issue is the seamless integration of all functionalities and their extensive validation to ensure that the entire scheme is without any flaw.

A typical block diagram of the total methodology for the design and validation of DOL wind biasing scheme is illustrated in Fig. 7.18.

The extensive measurements of winds over a period on the launch base are needed, and using this data the nominal and 3σ variation of the wind for each month is to be generated. The wind variation from the wind measured before launch and the wind prevailing at launch time is also to be assessed by measuring the winds for a few days prior to the launch date as well as just a few hours before flight. Generally this variation is not significant, and thus DOL wind biasing trajectory allows all weather launches and totally eliminates the risk of launch postponement.

Figure 7.19 shows a typical plot of $Q\alpha$ vs time during the atmospheric phase of a flight: (a) without wind biasing, (b) seasonal mean wind biasing and c) DOL wind biasing trajectory. The advantage of using DOL wind biasing is quite evident and ensures benign load conditions to the vehicle during the flight.
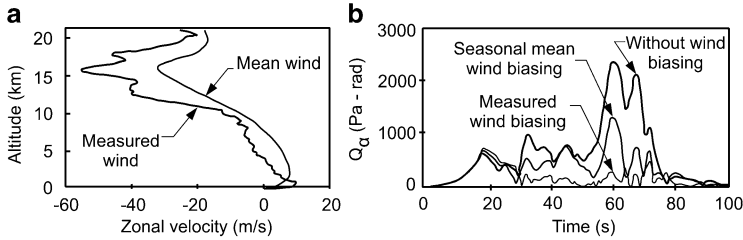
Fig. 7.19 Effect of Day-of-Launch wind biasing (**a**) Wind (**b**) Aero load indicator

7.9   Thermal Constraints

Another important mission requirement is to ensure that sensitive subsystems of vehicle and payload do not experience excessive thermal loads during the flight. The separation altitude of the payload fairing is to be carefully decided during trajectory design, and the basic guideline used is that the heat flux experienced by the payload has to be less than 1135 W/m^2 (1 sun heat), which is considered to be safe. Since the flow regime in this transition changes from continuum to free molecular the estimation of heat flux has to be carried out in detail and the separation altitude to be confirmed as safe.

For upper stages with low thrust-to-mass ratio and with a flight path angle very close to the local horizontal direction at stage ignition, the trajectory altitude may decrease or dip during certain segment of the flight before the vehicle gains enough acceleration. In such cases trajectory needs to be designed with restriction on the extent of dipping of altitude to ensure that the vehicle does not experience unsafe thermal loads.

These types of constraints are highly non-linear and very sensitive to changes in the variables in the initial and middle portions of the trajectory. The thermal boundary poses a critical constraint for the vehicle, and in this region it is better to use a flat trajectory with a limit on flight path angle. As the thermal environment is function of atmospheric density (altitude) and relative velocity, the allowable thermal boundary can be defined in terms of altitude-relative velocity space. During flight with sensitive systems exposed to ambient, the vehicle trajectory has to be in the safe zone. A typical profile of vehicle velocity and altitude indicating the safe thermal boundary is given in Fig. 7.20.

It is essential to define the minimum dip altitude based on thermal considerations and use it as a constraint during the trajectory design. The flight path of the upper stage is to be shaped such that the altitude does not dip beyond specified limits. Although the trajectory dip may help in maximizing the payload, the STS mission trajectory has to be designed such that the constraint on heat flux has to be strictly followed to protect the payload from the thermal considerations.

Fig. 7.20 Thermal
boundary profile
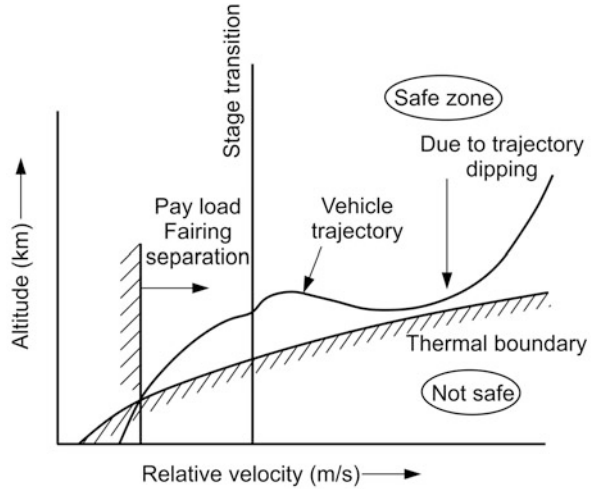
7.10   Mission Sequence Design

Mission sequence design involves defining optimum sequence of launch events for the STS mission starting from T_0, first-stage ignition time, till the satellite injection into orbit, satisfying all mission constraints. Mission sequence plays a vital role in achieving the required mission performance of STS while satisfying the vehicle, ground systems and subsystem requirements during its mission. Suitable mission sequence is essential for the smooth functioning of the vehicle systems and to achieve the defined mission successfully. To ensure this all the flight events in terms of engine ignition, vertical rise, pitch/yaw manoeuvre, engine shut-off, stage separation, payload fairing separation, coast phase, terminal stage engine shut-off and satellite separation, etc. take place as planned for the nominal as well as off-nominal flight environments. The optimum sequence of events valid for one flight environment may not be safe for another environment. Therefore, to meet all the system requirements, it is essential to define the event sequence in real time by vehicle on-board systems, depending on the flight environment. The optimum strategy is to detect the critical events in vehicle on-board, i.e. real-time decision (RTD) and further events sequencing are referenced with respect to the RTD till the next critical event is identified. This process starts from the first-stage ignition and continues till satellite injection.

7.10.1   Mission Sequence Strategy

Mission sequence strategy for a typical three-stage STS system is given in Fig. 7.21. First-stage ignition command is issued by ground checkout system. The entire
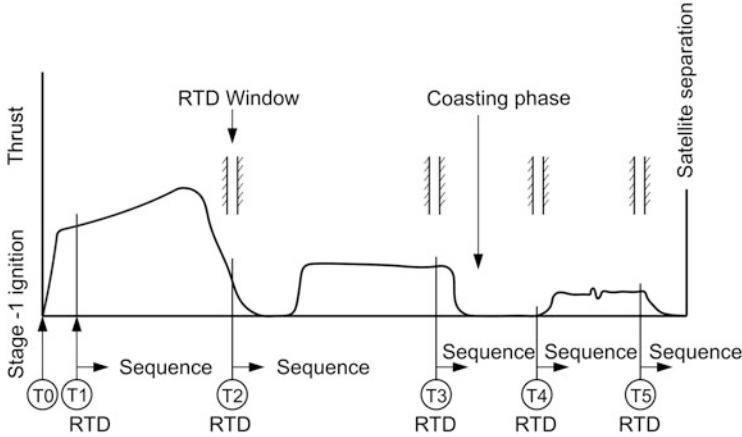
Fig. 7.21 Typical mission sequencing

vehicle sequencing is activated only after confirming the positive vehicle lift-off from launch pad which is linked to unambiguous connection de-mating between ground and vehicle and termed as last-minute plug (LMP) pullout. This RTD is identified as T1 in Fig. 7.21, and all subsequent events are linked to T1. It is possible to detect this event either by a sit-on-connector or by monitoring the chamber pressure of the booster motor. Appropriate values are to be fixed for these parameters considering all possible scenarios for absolute detection of T1.

Regarding the real-time decision between stage transitions, the longitudinal acceleration of the vehicle available in the inertial navigation system is generally used to detect the burnout of the stages. Once the propulsion stage thrust starts falling during the end of its burn (tail-off region), the acceleration starts dropping. Suitable value/threshold for acceleration sensing in each stage can be decided for detecting the tail-off based on clean separation requirement. This raises a flag in the on-board computer, which, in turn, initiates the RTD events for stage separation. This RTD is represented as T2 in Fig. 7.21. The value of the threshold for RTD sensing can be set considering various factors like the burn-time dispersions for the motor, tail-off dispersions, subsequent events during the tail-off region, etc. The detection of RTD has to be highly reliable and should not give false detection during any phase of flight. Therefore the redundant values of the longitudinal accelerations available in inertial navigation system are used for RTD with proper selection logic. Once the detection of real-time decision is available on board, all subsequent events of sequencing are linked to the new RTD timing till the next RTD detection.

In certain cases, once the instantaneous impact point of the vehicle reaches the allowable range safety boundary, it is essential to shut off the stage and separate it,

Table 7.1 Typical Event Sequencing

Event no.	RTD	Criteria for RTD	Time	Event
1	–	–	T0	Stage-1 ignition
2	T1	LMP pull out	T1 + 0	LMP pull out
3			T1 + x1	Stage-1 control on
4	T2	Acceleration ≤ a	T2 + 0	RTD sensing
5			T2 + x2	Stage-1 control off
6			T2 + x3	Stage-1 separation
7			T2 + x4	Stage-2 ignition
8			T2 + x5	Stage-2 control on
9			T2 + x6	PLF separation
10	T3	Instantaneous	T3 + 0	Stage-2 shut-off
11		Impact point > limit	T3 + x7	Stage-2 control off
12			T3 + x8	Stage-2 separation
13			T3 + x9	Coast control on
14	T4	Desirable position, velocity	T4 + 0	Stage-3 ignition
15			T4 + x10	Coast control-off/
				Stage-3 control on
16	T5	Target orbit	T5 + 0	Stage-3 shut-off
17			T5 + x11	Stage-3 control off
18			T5 + x12	Stage-3 coast control on
19			T5 + x13	Stage-3 coast control off
20			T5 + x14	Satellite separation

to meet the mission safety requirements. This point is decided based on the vehicle reaching specified position and velocity vector. This RTD is marked as T3 in Fig. 7.21. All further sequencing is based on T3.

If the vehicle mission has the feature of coasting between two stages as shown in Fig. 7.21, it is essential to determine optimum time for igniting the next stage, represented as RTD T4. Further sequencing in this stage is referenced with respect to T4.

Once the mission target conditions are achieved, it is essential to shut off the final-stage engine. This is decided in the vehicle on-board by the closed-loop guidance algorithm and is represented as RTD T5 in Fig. 7.21. Vehicle sequencing beyond engine shut-off till satellite separation are linked with T5.

The sequencing events which are linked with RTD should not be triggered inadvertently (false alarm) at any other time than intended. This is taken care by defining appropriate time window for each RTD, by taking into account all possible dispersion scenarios of propulsion systems. In case of the failure in RTD identification, due to malfunctioning of any of the elements in the chain, the window-out is taken as default time, and rest of the sequencing events are linked with this time. Typical event sequencing based on RTD is given in Table 7.1.

7.10.2   Typical Stage Transition Sequence

Stage transition is a crucial event for STS mission. A typical event sequencing design strategy is explained below. There are three major conflicting requirements during stage transition: (1) performance, (2) clean separation and (3) vehicle controllability.

For a clean separation, the thrust of the spent stage has to be as low as possible. For the case of solid motors, this event can occur much later than motor burnout time. Also, for both the cases of solid motors and liquid engines, there can be large dispersions in the action time. Therefore, if the sequencing has to be based on time, to ensure minimum thrust for both the cases of shortest and longest burn time, one has to wait for long time for separating the spent stage even after motor burnout. Only after the separation of the spent stage the next stage can be ignited. The long gap between lower-stage burnout and upper-stage ignition has severe impact on the performance.

Regarding the vehicle controllability, it is essential to switch off the control in the spent stages when the thrust level is sufficiently high to avoid the control-induced disturbances. After control switch-off there is a time gap for stage separation, next stage ignition and control on from the next stage. This introduces a large no-control zone between the stage transitions. This can cause the increased rate and error, and in certain cases, the upper stage may not be able to capture the vehicle with such large error, finally ending up with mission failure. Therefore, to avoid such scenario, it is essential to reduce the no-control zone. For the cases of fixed time-based sequencing, the minimum thrust requirement for control jeopardizes the mission if there is large dispersion in the spent stage motor performance.

Because of these conflicting requirements, it is not possible to fix the time-based sequencing a priori for such crucial events. Thus it becomes absolutely essential to determine the burnout of each propulsive stage in real time on-board by utilizing proper sensors like pressure sensors measuring the motor chamber pressure or longitudinal acceleration using accelerometers in the navigation system. Based on the output of these sensors, RTD can be made by the on-board computer to find the time at which the stage has burned out, thrust requirement for control off and the thrust sufficiently small for safe stage separation. This is done in real time by the on-board computer. All further sequencing is fixed based on the RTD event as in Fig. 7.22. Suitable window-in and window-out times are fixed for the RTD as explained earlier. A typical event sequencing after the RTD considering the expected dispersions on the upper stage is explained in Fig. 7.22.

7.11   Propellant Loading Requirements

The propellant mass required in each stage to achieve the specified mission is decided during configuration design phase. This is one of the major requirements for the design of the propulsion systems along with performance parameters. One of
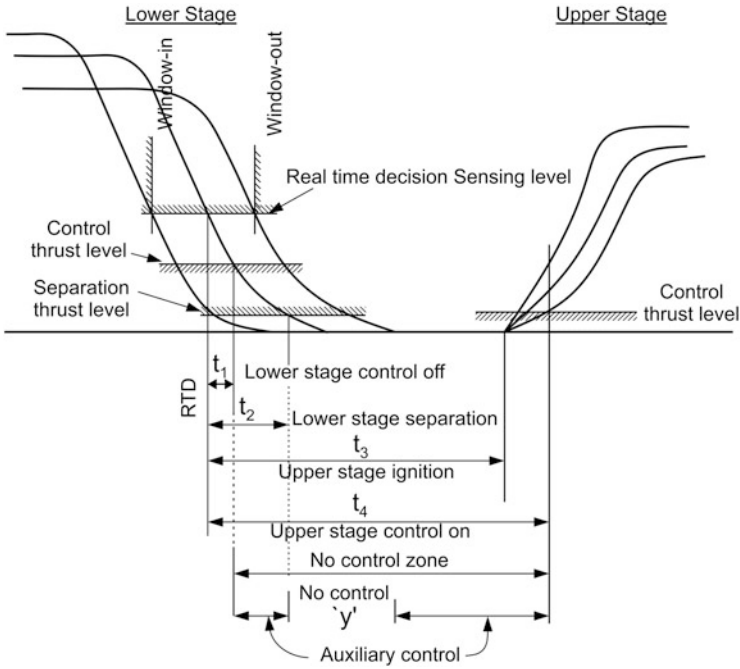
Fig. 7.22 Typical stage transition sequencing

the important performance parameters for the liquid propulsion system is the mixture ratio. Maximum performance in terms of specific impulse is achieved at an optimum mixture ratio, depending on the type of propulsion system and design specification. The propellant tanks are designed to accommodate the propellants which need to be consumed at the specified mixture ratio. Sometimes the optimum mixture ratio demands large tank size for one of the propellants, which in turn has the impact on the mission performance. Therefore, trade-off studies considering the improvement in specific impulse and reduction in structural mass have to be carried out to arrive at a suitable mixture ratio which gives the maximum performance. The details are explained in Chap. 9.

During flight, depending on the flight environment, there can be deviation in achieved mixture ratio. If it is more than the specified value, the oxidizer is consumed at a faster rate. Once the oxidizer depletes, the engine thrust comes down and the burnout of the stage happens. Under such conditions, to ensure the safety of the engine, it is essential to shut off the engine. Therefore, the effective functional duration of the stage becomes less than expected, and the total impulse imparted by the stage is also less. At the time of burnout, unused fuel Δm_{fu} remains in the tank as shown in Fig. 7.23.

In order to avoid such scenarios, generally either active or passive mixture ratio control systems are introduced in liquid propulsion systems, which ensure that the mixture ratio is close to the nominal even in the dispersed flight environments.

Fig. 7.23 Propellant
consumption history pattern
for different mixture ratios
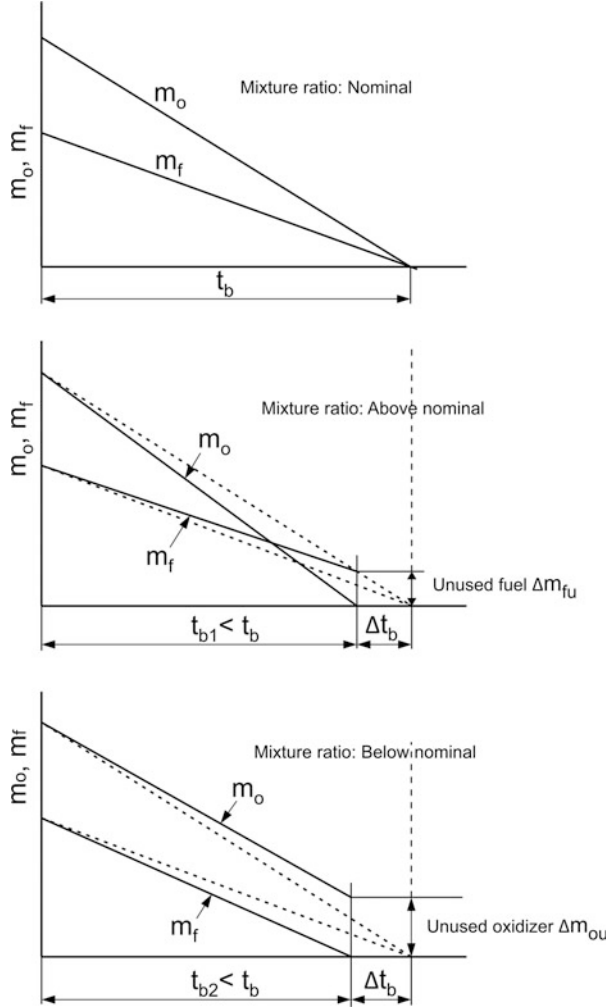
However, due to the errors in sensors used for the mixture ratio control as well as
due to the deviations in the subsystems used with respect to the predicted values,
there can be small dispersions in the realized mixture ratio. The deviated mixture
ratio and the propellant depletion before the expected time have the following
implications: (1) reduction in the active functional duration of the stage, (2) tail-off
characteristics.

   The reduction in the burn duration has a significant impact on the mission
performance. The different tail-off characteristics due to the different types of
depletion cause severe environment to the vehicle subsystems and satellites. The
tail-off characteristics and frequency contents of pressure and thrust fluctuations are
different for the various types of depletion, viz. (1) simultaneous depletion,

(2) oxidizer depletion, (3) fuel depletion, (4) oxidizer followed by fuel depletion and (5) fuel followed by oxidizer depletion. Depending on the vehicle system characteristics and tail-off characteristics, the impact of the tail-off characteristics on the health of the subsystems are analyzed, and based on the results, the specified depletion characteristics are chosen. This is achieved by (1) tuning the valve parameters to ensure the specified depletion characteristics occur in flight and (2) loading the propellants (oxidizer and fuel) such that the specified depletion occurs. This is termed as propellant loading analysis in mission design process.

To achieve the specified depletion under all the possible environments of parameter dispersions including mixture ratio, the required loading can be 'skew-loading', i.e. one of the propellants has to be loaded much beyond that corresponding to the nominal mixture ratio. This extra loading has severe performance impact. Therefore, to get more favourable performance, the loading is finalized based on Monte Carlo (MC) analysis. In the propellant loading MC analysis, a detailed propulsion system model is used considering all performance parameters and other vehicle systems and the interfaces among them. Using this model, for different propellant loading combinations, MC analysis is carried out, perturbing various performance parameters including mixture ratio which gives the probability of each type of depletion and the vehicle performance. Using these results, the propellant loading combination within the capacity of propellant tanks is to be selected. This has to meet the ullage volume requirement which gives the desired depletion while meeting the required performance. This propellant loading has to be used in the mission design process and implemented in flight.

7.12   Velocity Reserve Requirements

The propellant loading requirement computed as part of configuration design is mainly to meet the equivalent velocity as demanded by a specific mission and skewed loading to meet the required depletion characteristics. In reality, during flight, the vehicle performance can have dispersions with respect to the predicted parameter values. In addition, due to the in-flight disturbances caused by the external and vehicle internal sources, there can be tracking error in the vehicle attitude with respect to the desired attitude as computed by the vehicle closed-loop guidance system. Under such environments, the energy provided by the propulsion system designed for the nominal vehicle performance may not be sufficient to achieve the mission target. Therefore, it is essential to provide extra energy in the vehicle propulsion system to take care of such in-flight uncertainties. The extra energy to be provided in the vehicle on-board is called 'velocity reserve' or 'guidance margin'.

If the mission is planned to take care of all the possible disturbances in the additive sense, then margin required can be very large. If such huge margin is to be built into the system, then the available energy for nominal performance is restricted, which has direct impact on the vehicle performance. It is to be noted

Table 7.2 Equivalent velocity reserve required

Sl No.	Parameter uncertainty	Incremental equivalent velocity required to achieve the target
Stage-1 (solid)		
1	I_{sp}	ΔV_1
2	Action time	ΔV_2
3	Propellant mass	ΔV_3
4	Structural mass	ΔV_4
Stage-2 (Liquid)		
5	I_{sp}	ΔV_5
6	Mixture ratio	ΔV_6
7	Oxidizer loading	ΔV_7
8	Fuel loading	ΔV_8
9	Chamber pressure	ΔV_9
10	Structural mass	ΔV_{10}
Stage-3 (Liquid)		
11	I_{sp}	ΔV_{11}
12	Mixture ratio	ΔV_{12}
13	Oxidizer loading	ΔV_{13}
14	Fuel loading	ΔV_{14}
15	Chamber pressure	ΔV_{15}
16	Structural mass	ΔV_{16}
17	PLF mass	ΔV_{17}
18	Satellite mass	Satellite mass

that the chances of all the performance parameters deviating simultaneously to create worst-case environment is very remote. Therefore, the guidance margin requirement is arrived at based on probability analysis. This is done in two ways: (1) Root Sum Square (RSS) method and (2) Monte Carlo Analysis method.

In RSS method, after the initial mission design for nominal vehicle parameters, each parameter is perturbed to its specified 3σ dispersion level. The impact of this dispersion is analyzed and extra equivalent velocity required to achieve the mission target under this dispersed environment estimated. The study is repeated for all the parameter dispersions, considering one at a time.

As an example, consider a three-stage vehicle with solid first stage, liquid second and third stages and assuming third stage is commanded to cut-off, the typical dispersion parameters are about 18. Their details and equivalent velocity variation due to each of these parameters are given in Table 7.2.

From data given in Table 7.2, the RSS of incremental equivalent velocity ΔV_m is computed as

$$\Delta V_m = \sqrt{\sum_{i=1}^{18} \Delta V_i^2} \tag{7.5}$$

The ΔV_m given in Eq. (7.5) is the velocity reserve required in the vehicle. Assuming ΔV_m has to be provided by third stage, considering the propulsion parameters of that stage, the extra propellant Δm required to provide ΔV_m is loaded additionally or kept as reserve, and this margin is called guidance margin.

In the MC analysis method, after the initial mission design with the nominal vehicle parameters, integrated mission simulation with detailed model as explained in Chap. 8 is carried out in MC mode. For all the specified dispersion parameters, type of dispersions, their specified distribution, etc. are analyzed. Each time, a set of parameters is selected assuming random variation on each of the parameters with the specified dispersions, and with the selected set, simulation is carried out. Such simulations are repeated with huge numbers of runs simulating that many sets of random values for the selected parameters. From each run, the incremental equivalent velocity required is computed. The total number of simulations n (say) depends on the convergence of the results. The n numbers of ΔVs are statistically analyzed to get the mean and 3σ of required ΔV. Then, depending on the defined success criteria, the required ΔV is finalized. This value can then be used to compute the required Δm as explained earlier. Once the Δm is loaded to the vehicle additionally or kept as a margin, then during STS flight, the probability of achieving the nominal targeted mission is same as the one used for designing Δm.

7.13   Satellite Injection Requirements

The satellites are required to be injected into the defined orbit with the following additional conditions: (1) rates at separation to be limited to the capabilities of the satellite control system, (2) sufficient distance between the separated stage and satellite is ensured and (3) attitude is achieved as desired by the functional requirements of satellites.

To meet the above requirements after the final stage cut-off, sufficient time is given before satellite separation to ensure that the residual thrust of the vehicle propulsion stage is zero. The satellite separation system is designed to provide the necessary differential velocity so that the distance between the satellite and separated stage is progressively increasing. The control system of the vehicle is designed such that, during the combined flight phase after cut-off, the vehicle is controlled to achieve the attitude as defined by the satellite requirements and the vehicle rate less than the specified limits. After achieving the required injection conditions, the satellite is injected into the orbit as represented in Fig. 7.24. To meet the specified distance between satellite and separated stage, satellite injection can be done along the orbital plane or in the out-of-plane direction. To avoid collision in

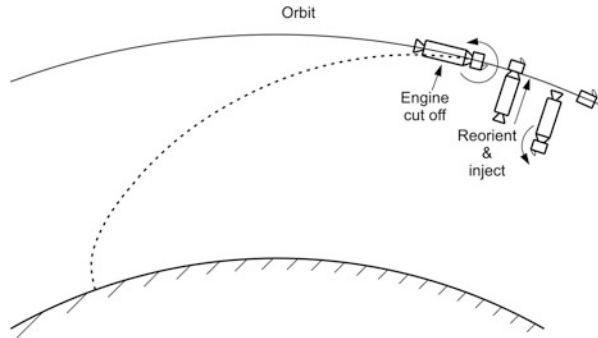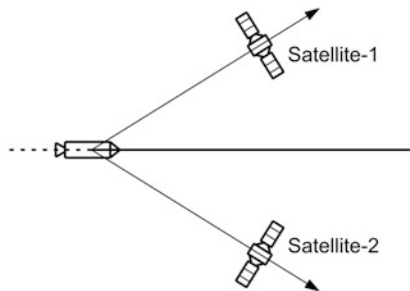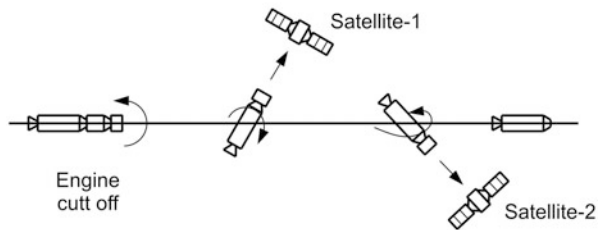**Fig. 7.24** Satellitere
orientation requirements

Fig. 7.25 Multiple satellite
injection requirements

(a) Distance requirement for multiple satellites

(b) satellites orientation requirement

the case of inadvertent re-ignition of the separated stage, generally, the separated stage is oriented in a direction away from the satellite motion.

In certain missions there are requirements for the launch of multiple satellites. In such cases it is essential to ensure that these satellites after separation continue to move in defined orbits without having any collision between any of the satellites during the entire period of their life as represented in Fig. 7.25a. This is achieved by defining proper sequence of satellite separation timings and reorientation requirements as given in Fig. 7.25b. The selection of separation velocities and flight sequence for the spacecraft are chosen such that they move in different but close orbits. It is also necessary to carry out long-term relative orbital motions for these

Fig. 7.26 Relative motion
of bodies after separation
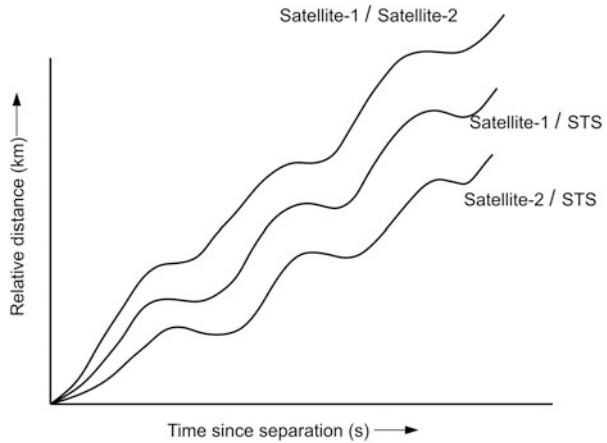
spacecraft. The possibility of plume interaction during the operation of reaction control thrusters and also during the passivation of terminal stage are to be analyzed in detail to avoid the potential damage to any of the spacecraft.

The collision possibility of any of the spacecraft or the final vehicle stage has to be studied considering the nominal and off-nominal performance of the final stage. The possible dispersions on separation system parameters are also to be considered. The orbital elements of the final stage and all spacecraft at respective separated conditions are to be estimated and relative distance movements of the objects at different time instances after separation analyzed. A typical analysis showing a typical relative motion between two spacecraft missions is shown in Fig. 7.26. From Fig. 7.26, it can be seen that the relative distances with respect to time are monotonically growing.

The relative minimum distance between various bodies has also to be assessed using Monte Carlo studies taking into account various dispersions like the left-out propellant in the final stage, separation velocity, the reaction control thruster variations, etc.

7.14   Propulsion Stage Passivation Requirements

The final stage of all launch vehicles enter the orbit, and as per the international guidelines on space debris it is mandatory to deplete all left-out propellants in a planned manner and the stage is put in a passive mode. This operation is termed as passivation of the stage. The quantity of left-out propellant in the stage depends on the propulsion performance of earlier stages. The requirement for passivation of the terminal stage is to initiate this action immediately after the satellite separation. But this can lead to contamination of the spacecraft. Therefore sufficient time gap has to

Fig. 7.27 Stage passivation
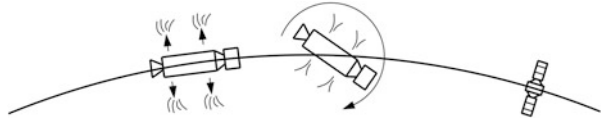
be provided between the separation of satellite and the initiation of the passivation as shown in Fig. 7.27.

The effective method for passivation is to introduce suitable vent nozzles. In order to avoid the reaction on the vehicle doublet-type nozzle configuration is generally chosen. There are several schemes for venting the trapped propellants, and their details are available in the references quoted. The vent circuits are prone to ice formation due to freezing, and this can lead to blockage or sometimes explosions. These issues are to be tackled during the design phase to confirm the smooth operation of venting as planned. Additionally one has to verify the lowest temperature attained and also the ice formation during the process to avoid failure of the system. To ensure high reliability, experimental evaluation of the thrusters in a high-altitude test facility using an identical scheme on ground has to be carried out.

7.15   Vehicle Tracking Requirements

Radio visibility of the vehicle during its entire mission duration from ground station tracking network is required for real-time monitoring of the flight to obtain the telemetry data from the flight for post-flight analysis. There is also requirement for issuing the tele-command from the ground whenever it is essential. The important parameters needed for vehicle tracking during the flight from a defined ground station are given in Fig. 7.28.

During trajectory design all the visibility parameters from a tracking station, namely, elevation, slant range, aspect angle, antenna azimuth, range rate and aspect angle, have to be estimated. Range information is needed to decide about the signal strength. Aspect angle is used to determine the loss in signal due to rocket exhaust coming in the path of the signal. During acquisition of signal (AOS) at least 5° elevation and during loss of signal (LOS) 2° elevation angle are desirable. However, a minimum of 2° elevation angle from AOS to LOS generally gives satisfactory visibility. For a better visibility, aspect angle has to be more than plume angle.

Therefore, even though there is good visibility in terms of elevation, when there is plume angle more than aspect angle as shown in Fig. 7.29, it is essential to plan an alternate ground station to acquire the telemetry data. If a taller or wider structure is present in front of the tracking station then for some combinations of elevation and antenna azimuth, the structure may block the passage of signal. In such cases, visibility has to be improved by positioning a mobile terminal at a vantage location or by redesigning the trajectory with implication on the payload and performance.

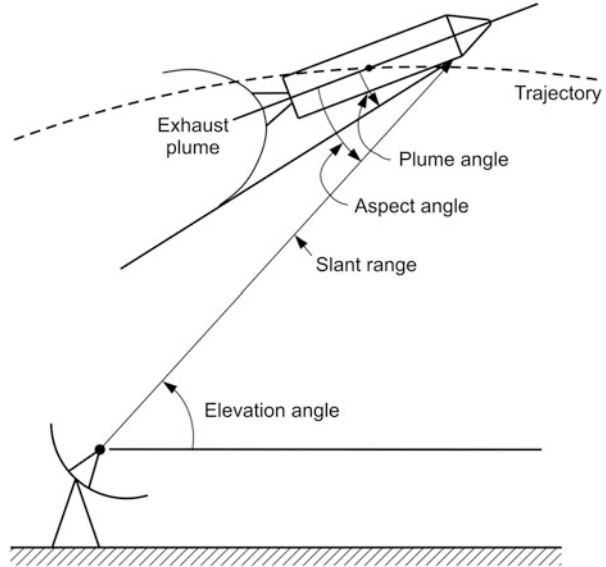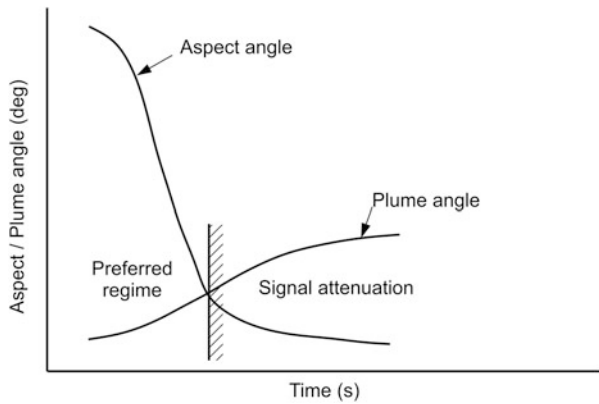**Fig. 7.28** Tracking
parameters

Fig. 7.29 Aspect and
plume angles

Further, the visibility coverage is needed for a certain length of time after the final
stage separation and spacecraft injection for monitoring the passivation and
re-orientation manoeuvres of the separated final stage. For preliminary orbit deter-
mination (POD) of the spacecraft, range, range rate, elevation and aspect angle
information are required.

   The tracking station locations are to be planned, depending on the mission, in
such a way that all critical events like stage ignition, stage burnout, stage separa-
tion, strap-on separation, payload fairing separation and spacecraft injection are
visible at least from one of the tracking stations. Continuous coverage is essential
except for long coasting phase where some gap in visibility may be acceptable.
Hence tracking stations are to be planned such that adequate overlap of coverage
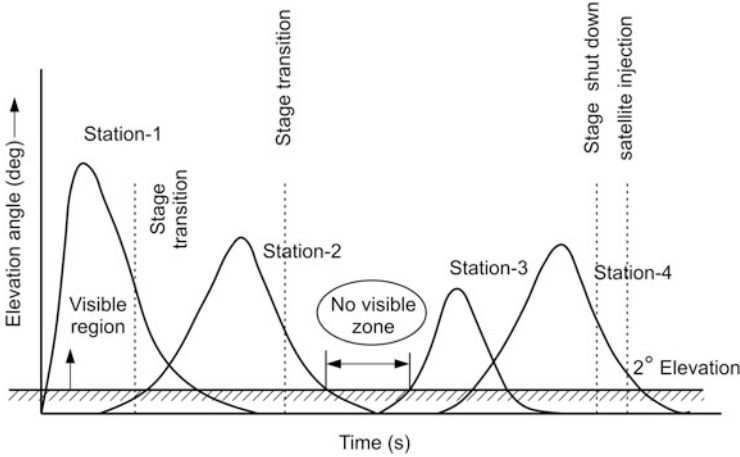
Fig. 7.30 Elevation angles of typical STS mission

from successive stations exists. Elevation angles from four ground stations of a typical STS mission with three stages are represented in Fig. 7.30. In case there is no visibility from any of the specified ground stations as shown in Fig. 7.30, the trajectory should be designed such that there are no critical events during that segment. This might have some implication on the payload that can be carried. For such regions where visibility does not exist and no critical events occur, data storage and delayed transmission has to be planned. In case of trajectories wherein critical events occur during the flight and the visibility from ground-based tracking stations is not available one has to necessarily plan the ship-based tracking during these periods.

7.16   Range Safety Constraints

The spent stages of launch vehicle are to be impacted either in safe land zones belonging to the country from which launch is originated or in safe international waters limited by specified boundaries. The mission design has to observe these conditions very strictly. Selection of a suitable launch azimuth depends not only on the final mission considerations but also on the safety aspects of the launch even if it has a penalty on the payload mass. At low altitudes the ground and instantaneous impact point (IIP) traces of the launch cannot violate the defined international boundaries.

The IIP trace of the vehicle is decided by the instantaneous position and velocity vectors as represented in Fig. 7.31. IIP is the point of impact of the vehicle on the Earth's surface in case the vehicle power goes off instantaneously. The separated stage impact point is the IIP corresponding to separation instant. Therefore, during
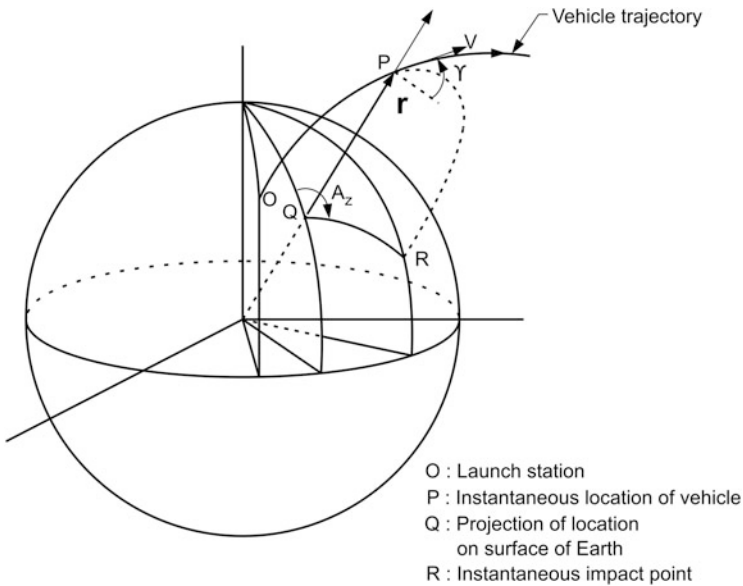
Fig. 7.31 Instantaneous impact point (stage impact)

mission design process, the position and velocity vector profiles have to be suitably designed to ensure that the range safety constraints are not violated.

During the mission design, another important study is the assessment of the hazards and risk levels to life and property in case of malfunction/failure of the vehicle during the ascent phase of the trajectory. The risk has to be assessed both during the lift-off phase involving the launch base facilities and during the subsequent phase of the flight involving the down range land masses. Therefore detailed studies are needed on vehicle failure modes, its impact on the trajectory and the land impact probability. The damage potential also depends on the type of failure and time at which failure occurs during flight. All these aspects have to be studied in detail during the initial mission studies. These data have to be appropriately utilized during launch to decide on the destruction of stages from range safety considerations.

7.17 Integrated Trajectory Design

Once the vehicle configuration is defined and mission specifications and constraints are identified, the integrated trajectory of the vehicle during ascent phase has to be suitably designed to meet the final satellite injection conditions in an optimum manner, while satisfying all the specified requirements and constraints. The trajectory design process involves generation of optimum trajectory profile, i.e. position and velocity histories from lift-off till satellite injection such that the end conditions

meet the required orbit. The selected trajectory has to inject maximum payload for the defined propulsion stages or maximize the reserve fuel (guidance margin) for the defined payload.

The vehicle trajectory in simple terms can be specified thus: the vehicle rises vertically at launch pad and then is made to orient its path smoothly on a continuous basis to achieve the required flight path angle at the specified altitude and velocity conditions for insertion of payload into the orbit. The vehicle has conflicting requirements like, on the one hand, moving vertically as far as possible to get out of the atmosphere very quickly and, on the other hand, to turn the vehicle as early as possible to attain the required flight path angle at relatively lower velocity regimes to avoid the steering losses in vehicle velocity. A trade-off study has to be made during trajectory design to ensure minimum losses due to drag and due to steering and to meet other constraints of thermal vehicle loads and range safety and tracking.

7.17.1 Trajectory Design Considerations

The main objective of trajectory design is to maximize the vehicle performance. During the integrated trajectory design process, various mission and vehicle subsystem requirements and constraints have to be carefully considered. The requirements are summarized as given below:

1. Launch azimuth has to be selected depending on the requirement on orbital inclination. For low inclination orbits, eastward launches are preferred whereas for high inclination orbits such as Sun-synchronous polar orbits, southward launch is desirable. But designer has to select only those launch azimuths which are declared safe throughout the flight path of the vehicle. The range of allowable azimuth is termed as the safe launch corridor.

2. Unlike a sounding rocket that is launched from a launch lug at an elevation, the launch vehicle which is resting on the launch pad is to be launched vertically. It ascends vertically for a given time to avoid the collision of the vehicle with the launch tower even under all combinations of disturbances. During the ascent, it is necessary to minimize the jet plumes from the vehicle impinging on the umbilical tower to avoid the damage to the tower. The impact of vertical rise time on payload is also to be considered.

3. The vehicle roll is facilitated to coincide with the pitch plane with the launch azimuth immediately after the initial lift-off and before any other manoeuvres are carried. This becomes essential in vehicles where the pitch axis of the vehicle is not aligned with the launch azimuth due to constraints of launch tower.

4. Subsequent to this the vehicle starts pitching down optimally. The upper limit on the pitch rate is dictated by the control system, vehicle inertia and tracking error. The lower limit is driven by the need to move away from the launch pad

as soon as possible to ensure safety of the range in the vicinity of the launch pad.

5. Once this phase is over, the gravity turn to the vehicle is initiated. This causes a transition in pitch and yaw rates from optimal values to the values which are needed to minimize the angle to attack to near zero.

6. Gravity turn is initiated based on altitude, transonic Mach number, aerodynamic load indicator values and payload obtainable. The trajectory is initially designed for seasonal mean wind profiles and follows gravity turn during the atmospheric portion till the closed-loop guidance is initiated.

7. For the vehicles with strap-on motors, their ignition and separation sequences have to be decided based on the optimum payload capability, peak dynamic pressure and acceleration at peak dynamic pressure for nominal and off-nominal trajectories. The dynamic pressure, angle of attack and body rates have to be kept small at strap-on separation.

8. Peak dynamic pressure is an important parameter from aerodynamic and structural considerations. If the peak dynamic pressure exceeds the prescribed limit, the trajectory has to be reshaped to obtain the specified values. It may be noted that large reduction is possible only by suitable design of thrust profile of the booster motor.

9. At burnout of stages, the dynamic pressure, angle of attack, bounds on attitude and attitude rates and also suitable design for the tail-off of the motors are to be ensured to meet the specified conditions. At the time of separation of each stage, unused propellant like injectant fluid used in secondary injection thrust vector control (SITVC) in the stage is jettisoned along with the inert mass, wherever possible.

10. Yaw manoeuvre, if necessary, is initiated after the atmospheric flight to avoid high angle to attack during the high dynamic and complex atmospheric flight regime. It is carried out generally from second stage burn phase in order to achieve the required inclination. Plane change is achieved during the ascent phase satisfying the constraints on instantaneous impact and planned stage impact points.

11. All the stage impacts have to be in safe zone, and instantaneous impact requirements with respect to flight over the land mass are also to be considered. Impact of spent stages is one of the difficult constraints which are encountered in trajectory design. Management of the impact of the spent upper stage in the safe zones is even harder since usually at that point the flight velocity is closer to orbital velocity. Hence for such cases, suitable orbit size has to be selected, and the trajectory has to meet the impact constraints even under off-nominal conditions.

12. An optimum unpowered duration of flight known as coasting can be employed to achieve the desired range, altitude and orientation whenever necessary and feasible. Thus the desired end conditions can be achieved without any expenditure on fuel. This segment can be between the powered flight portions of the same stage or two different stages.

13. The end constraints such as altitude and inclination have to be satisfied for circular orbits whereas for elliptical orbits perigee altitude, apogee altitude, arguments of perigee and inclination constraints have to be satisfied.
14. Suitable 'guidance margin' fuel has to be reserved as explained earlier.
15. 'No-control zones' between stage transitions are to be minimized during the steering design.
16. The $\pm 3\,\sigma$ variations of the critical parameters of the subsystems of the vehicle like propulsion systems, aerodynamics and navigation system are to be specified. Trajectory designer has to simulate these variations (like the off-nominal performance of motors, aerodynamic drag, etc.) and verify the performance. Designer has to ensure that the constraints are met to the extent possible under these variations.

7.17.2   Trajectory Design Strategy

It is to be noted that all the requirements and constraints explained above depend on trajectory (position and velocity vectors) profile. Therefore, in order to achieve the specified requirements, it is essential to modulate the trajectory profile as per the requirements of various phases of mission, and there must be suitable control variables required to modulate the trajectory. This is achieved as explained below:

Consider three-dimensional trajectory dynamics of the vehicle as explained in Chap. 8:

$$\dot{r} = V \tag{7.6}$$

$$\dot{V} = \frac{1}{m}\,[BI]\,[F_T + F_A] + G_I \tag{7.7}$$

where

r, V = Position and velocity vectors of vehicle with respect to ECI frame
m = Vehicle mass
$[BI]$ = Body-to-ECI frame transformation matrix, function of vehicle attitude θ, ψ, ϕ.
G_I = Gravity acceleration vector
F_T = Thrust vector of the vehicle
F_A = Aerodynamic force vector of the vehicle

The thrust vector F_T depends on the propulsion performance and atmospheric density. The atmospheric density is function of altitude, which in turn depends on r. Mass m is function of propellant depletion history, decided by the propulsion system used. The aerodynamic force vector F_A depends on the aerodynamic characteristics of the vehicle (C_N, C_S, C_A), dynamic pressure and angles of attack. Therefore, F_A can be assumed as function of vehicle shape and r and V. The gravity vector G_I is function of r. The body-to-inertial transformation matrix depends only

Fig. 7.32 Vehicle
trajectory dynamics
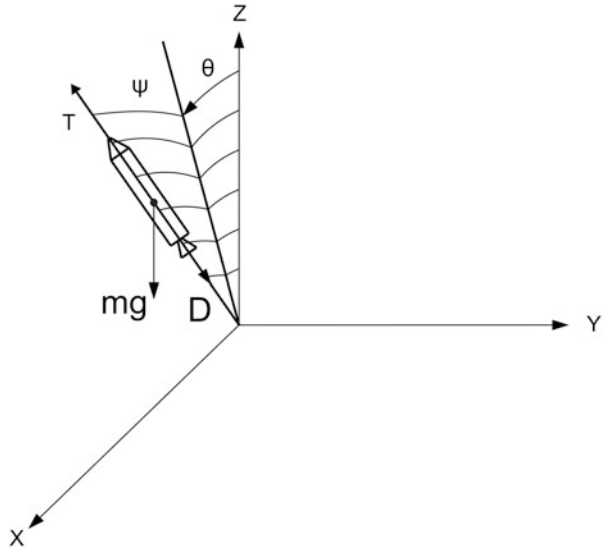
on vehicle attitude profile. Therefore, it can be concluded that for a defined vehicle, the trajectory dynamics is function of vehicle attitude.

For a simplified version as shown in Fig. 7.32, where thrust and velocity vectors along vehicle longitudinal axis (normal and side forces) are zero, the translational dynamics is given by

$$\ddot{X} = \left[\frac{T(T_V, x, y, z) - D(C_D, x, y, z, u, v, w)}{m} \right] \cos \Psi \sin \theta + g_x(x, y, z) \quad (7.8)$$

$$\ddot{Y} = \left[\frac{T(T_V - x, y, z) - D(C_D, x, y, z, u, v, w)}{m} \right] \sin \Psi + g_y(x, y, z) \quad (7.9)$$

$$\ddot{Z} = \left[\frac{T(T_V - x, y, z) - D(C_D, x, y, z, u, v, w)}{m} \right] \cos \Psi \cos \theta + g_z(x, y, x) \quad (7.10)$$

The vehicle position and velocity are given by

$$x = \int u \quad (7.11)$$

$$y = \int v \quad (7.12)$$

$$z = \int w \quad (7.13)$$

$$u = \int \ddot{X} \qquad\qquad\qquad (7.14)$$

$$v = \int \ddot{Y} \qquad\qquad\qquad (7.15)$$

$$w = \int \ddot{Z} \qquad\qquad\qquad (7.16)$$

From the above discussions, it can be concluded that for a given vehicle configuration with the specified propulsion stages, the trajectory is a function of vehicle attitude θ, Ψ and ϕ. By modulating θ, Ψ, ϕ the required trajectory can be achieved.

Therefore, the trajectory design process can be stated as the estimation of optimum attitude profile (steering profile) which generates the desired trajectory which maximizes the vehicle performance while satisfying various requirements and constraints.

The above problem can be stated mathematically thus:

Select control variable vector $\mathbf{c(t)}$ such that

$$\text{minimize} \qquad J = \phi(\mathbf{X}(t_f), t_f) \qquad \text{(performance index)} \qquad (7.17)$$

subject to

$$dX/dt = \mathbf{f}(\mathbf{X}(t), \mathbf{c}(t), t) \quad (\mathbf{X}(t) : \text{state variable vector}) \qquad (7.18)$$

$$\psi(\mathbf{X}(t_f, \mathbf{c}(t_f), t_f)) = 0 \quad \text{(boundary condition)} \qquad (7.19)$$

$$\mathbf{g_l} \leq g(\mathbf{X}(t, \mathbf{c}(t), t)) \leq \mathbf{g_u} \quad \text{(path constraints)} \qquad (7.20)$$

In the trajectory optimization process

$c = \{\theta, \Psi\}$

$X = \{x, y, z, u, v, w\}^T$

$\Psi = \{\text{orbital parameters at injection}\}$

$g = $ Inequality constraints such as range safety, tracking, heat flux, etc.

$\mathbf{g_l}$, $\mathbf{g_u} = $ Limits of the constraints

Therefore, it can be concluded that the trajectory design process is an optimal control problem. By suitable parameterization process, the trajectory design can also be considered as constraint non-linear optimization problem.

Typical equivalent parameters are given below:

1. *Control variables*:

    (a) θ and $\dot{\Psi}$ profile
    (b) $\dot{\theta}$ and $\dot{\Psi}$ history
    (c)   Staging duration
    (d)   Coasting duration
    (e)   Stage ignition in case of multi-stages

2. *Optimization variables*:

   (a) Maximization of payload
   (b) Minimization of burn time
   (c) Maximization or minimization of one of the orbital parameters

   The optimization parameter depends on the specific mission requirements:

3. *Boundary conditions*:

   (a) Required orbital conditions at injection

4. *Path constraints*:

   (a) Range safety constraints
   (b) Dynamic pressure constraints
   (c) Tracking parameter constraints
   (d) Thermal constraints

7.17.3   Integrated Trajectory Design Process

The integrated trajectory design is an iterative procedure. The central core of the process consists of vehicle flight mechanics and an optimizer. The input to the flight mechanics process is the vehicle data, vehicle sequencing and environment such as gravity, atmospheric properties and wind conditions. The input conditions for the optimizer are the mission requirements and vehicle constraints. The flight mechanics and optimizer interact iteratively, and the converged optimum trajectory is the output of the design process as explained in Fig. 7.33.

Using vehicle data, wind data and with initial estimates for the attitude programme and staging events such as coasting time, stage ignition and cut-off times, the flight mechanics model propagates the trajectory from lift-off till the initially defined cut-off time. During the trajectory propagation process, various events as defined such as vertical rise, gravity turn initiation, wind biasing with the defined wind profile, initiation of steering after gravity turn from the specified time, stage events, etc. are used. The propagated trajectory parameters are given as input to the optimizer. Using the trajectory parameters, mission-defined requirements and constraints, optimizer updates the steering angles, the coasting time, cut-off time and other design parameters to optimize the objective function while satisfying the constraints. These updated steering and control parameters are used in the flight mechanics to re-propagate the trajectory from the lift-off till the refined cut-off time and the trajectory outputs passed to the optimizer. This process repeats till the solution converges.

The converged solution gives the optimum steering programme and trajectory profile which achieves the optimum performance of the vehicle while meeting all the requirements and constraints including the wind biasing requirements.
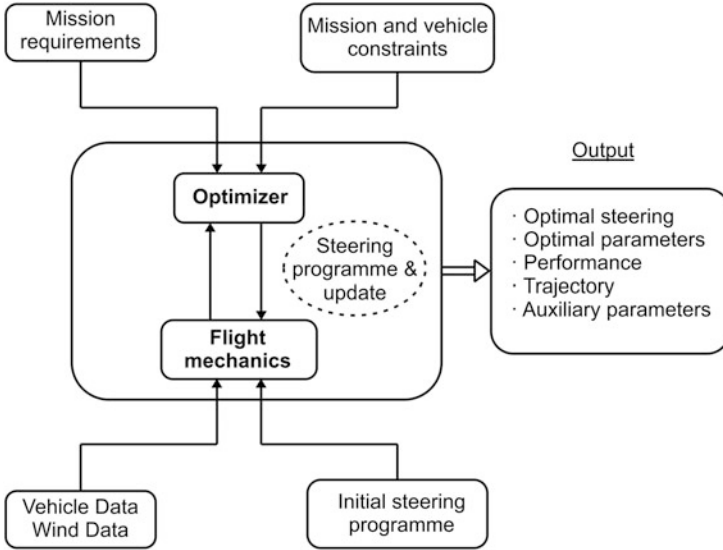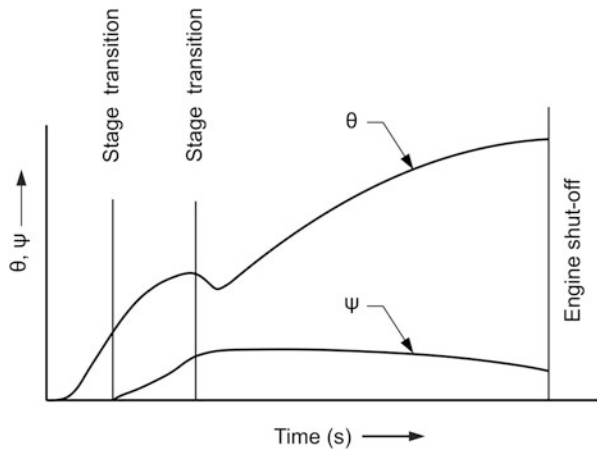
Fig. 7.33   Integrated trajectory design

Fig. 7.34   Steering profile

7.17.4   Optimum Mission Profile

The optimum steering program of a typical three-stage STS mission is given in Fig. 7.34. The altitude and inertial velocity profiles are given in Fig. 7.35. The flight parameters during atmospheric flight phase are represented in Fig. 7.36. A typical mission profile is given in Fig. 7.37.

   Three-dimensional point-mass trajectory dynamics as defined above has to be expanded to include the rotational motions of the vehicle about the centre of gravity
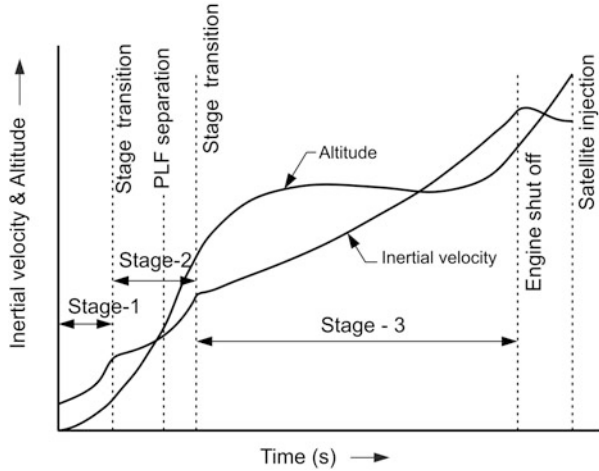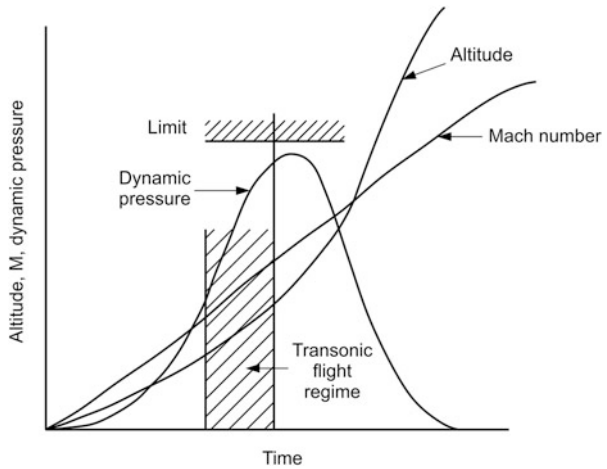
Fig. 7.35 Trajectory profiles

Fig. 7.36 Flight parameters during atmospheric flight phase

to suitable six-degrees-of-freedom trajectory dynamics. Once the basic trajectory meeting the overall (6-DOF) requirement is generated, the navigation, guidance and control (NGC) functions are progressively integrated into this 6-DOF trajectory dynamics to evaluate the detailed performance of the vehicle, subsystems and mission. Details of dynamics, their formulations, performance evaluation, etc. are given in Chap. 14.

The mission thus designed forms as reference input for all further subsystem design studies. These design aspects of subsystems are discussed in detail in the subsequent chapters of the book.
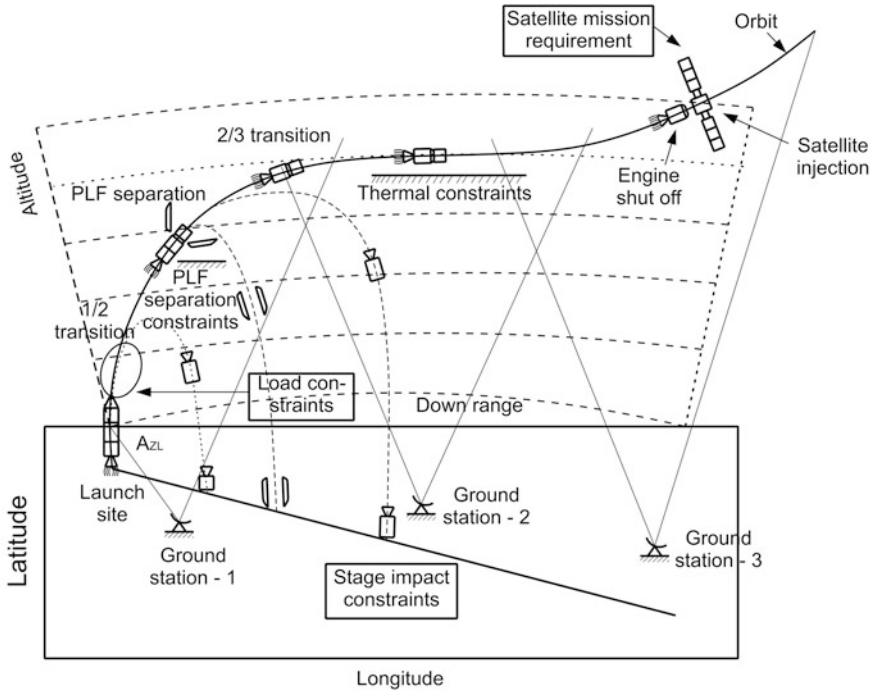
Fig. 7.37 Typical STS mission profile

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